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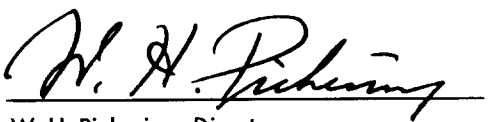
Preface

The Space Programs Summary is a six-volume bimonthly publication designed to report on JPL space exploration programs and related supporting research and advanced development projects. The titles of all volumes of the Space Programs Summary are:

Vol. I. *The Lunar Program* (Confidential)
Vol. II. *The Planetary–Interplanetary Program* (Confidential)
Vol. III. *The Deep Space Network* (Unclassified)
Vol. IV. *Supporting Research and Advanced Development* (Unclassified)
Vol. V. *Supporting Research and Advanced Development* (Confidential)
Vol. VI. *Space Exploration Programs and Space Sciences* (Unclassified)

The Space Programs Summary, Vol. VI, consists of an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of the JPL quality assurance and reliability efforts, and the environmental- and dynamic-testing facility-development activities; and a reprint of the space science instrumentation studies of Vols. I and II.

Approved by:



W. H. Pickering, *Director*
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I. *Mariner* 1967 Project

A. Introduction

The *Mariner* 1967 Project continues and extends the work of the previous *Mariner* Projects. One element of the Project is concerned with the Venus flyby mission in 1967, while the other two are concerned with the *Mariner IV* spacecraft now in space.

1. *Mariner* Venus 67 Project

The major task of the *Mariner* 1967 Project is the Venus flyby mission, named the *Mariner* Venus 67 Project. The following mission objectives have been established:

> *The primary objective of the Mariner Venus 67 Project is to conduct a flyby mission to Venus in 1967 in order to obtain scientific information which will complement and extend the results obtained by Mariner II relevant to determining the origin and nature of Venus and its environment.*
>
> *Secondary objectives are to acquire engineering experience in converting and operating a spacecraft designed for flight to Mars into one flown to Venus, and to obtain information on the interplanetary environment during a period of increasing solar activity.*

With the authorizing of this project late in December of 1965, and the launch opportunity occurring in June 1967, the period available to the *Mariner* Venus 67 Project for planning, design, test, and other flight preparation is 18 mo. Because of this minimum length of time, application, to the fullest extent possible, must be made of techniques developed from the experience of prior missions, and existing hardware must be utilized.

The single flight spacecraft, designated M67-2, is currently being prepared by converting the spare spacecraft (MC-4) that was built for the *Mariner* Mars 1964 Project.[1] An *Atlas-Agena D* launch vehicle will inject the spacecraft into a 120-day transit trajectory. The launch readiness date has been set at June 10, 1967, with a 14-day launch period starting June 12. Ranging provisions have been authorized for this mission which will offer the possibility of considerably enhancing the celestial mechanics experiment. At encounter, an unusually sharp perturbation will occur.

[1]Portions of the proof test model (PTM) spacecraft and spares from the MM64 Project are also being prepared as a flight-support spacecraft (M67-1); there will be no PTM as such. The flight-support spacecraft will serve the double function of pseudo-PTM and backup spacecraft for qualifying spare subsystems.

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Seven scientific experiments have been approved for the *Mariner* Venus 67 mission:

Experiment	*Principal investigator*
S-band radio occultation	Arvydas J. Kliore, Jet Propulsion Laboratory
Ultraviolet photometer	Charles A. Barth, University of Colorado
Dual-frequency radio propagation	Von R. Eshleman, Stanford University
Helium magnetometer	Edward J. Smith, Jet Propulsion Laboratory
Solar plasma	Herbert S. Bridge, Massachusetts Institute of Technology
Trapped radiation	James A. Van Allen, State University of Iowa
Celestial mechanics	John D. Anderson, Jet Propulsion Laboratory

The first experiment requires the use of only the RF transmission subsystem on the spacecraft, and the last one utilizes only the tracking doppler data received from the RF carrier. Of the remaining five experiments, four are to be accomplished with the existing instrumentation, with only minor modifications. Only the dual-frequency radio propagation experiment of Stanford University requires the incorporation of a new scientific instrument into the payload.

Other changes to the basic spacecraft design are necessitated by the fact that the spacecraft will travel toward, rather than away from, the Sun and also because conversions must be made to accommodate revised encounter sequencing and science payload. In particular, modifications are needed in the following areas:

(1) Scientific data automation system

(2) Antenna pattern and orientation

(3) Thermal control

(4) Solar panel configuration

(5) Planetary sensors

During the September–October reporting period, continued emphasis was given to the delivery of the remaining flight-quality telecommunications and science hardware. Once this phase has been successfully completed, the spacecraft environmental cycle will be initiated.

The first system test on the M67-1 spacecraft has been concluded. This test, however, was run without science and with some prototype engineering units installed.

Spacecraft system dual-frequency radio compatibility tests have indicated the possible extent of performance degradation. A plan to locate and isolate these interference sources is currently being developed.

Precision two-way doppler measurements were taken on the *Mariner IV* signal to test a new diplexer at the Mars Station. The received power levels at the Mars and Venus Stations were —174.4 and —176.3 dbm, respectively.

Coincident with the September solar flare, a 6 to 11% degradation of *Mariner IV* solar panel capability has been observed.

2. *Mariner* Mars 1964 Project

Residual activity of the *Mariner* Mars 1964 Project consists primarily of continuing analysis of tracking data, publication of results by the principal investigators, and preparation of final reports. However, periodic detection, recording, and commanding activities involving the *Mariner IV* spacecraft have been assigned to the Deep Space Network (DSN). Using elements of the experimental facilities at Goldstone Station and advanced R&D techniques, attempts are being made to communicate with the spacecraft. Results are published in *SPS*, Vol. III.

3. *Mariner IV* Project

In the second half of 1967, *Mariner IV* will again be within the normal communication range of the DSN stations. At that time an attempt will be made to obtain additional telemetry data from *Mariner IV*. The *Mariner IV* Project comprises the preparation, support, and actual reacquisition of the spacecraft.

The objectives, authorized in December 1965, simultaneously with those for the Venus flight, have been established as follows:

> *The primary objective of the Mariner IV Project is to obtain scientific information on the interplanetary environment in a region of space further from the Sun than the orbit of Earth during a period of increasing solar activity in 1967, using the Mariner IV spacecraft still operating in orbit around the Sun.*

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The secondary objectives are to obtain additional engineering knowledge about the consequences of extended exposure of spacecraft equipment in the interplanetary space environment and to acquire experience in the operation of a planetary spacecraft after a prolonged lifetime in deep space.

Flight operations on this project (providing that a functioning spacecraft allows this to occur) will overlap the *Mariner* Venus 67 Project flight activities.

B. Systems

1. Systems-Test-Complex Data System

A computer-based data system (STCDS) will be used to acquire, process, record, and display real-time data from *Mariner* Venus 67 spacecraft system tests [SPS 37-41, Vol. II (Confidential)]. Several significant STCDS development milestones were accomplished in the current reporting period. In early October the capability to acquire, record, and process spacecraft telemetry data was demonstrated, including the output of formatted data displays on line printers and teletype page printers. This processing includes the capability to suppress, on any data channel, the printout of all data samples whose value has changed less than some predetermined amount from the previously displayed value. Other capabilities provided are limit checking, conversion to engineering units, and the changing of tolerances and limits used in processing by card control in real time. Several routines for the special processing of telemetry were also demonstrated; calculation of system and subsystem power, command detector status, temperature measurement corrections, and the decommutation and formatted display of science telemetry, with the exception of the encounter telemetry data which is reproduced from the tape recorder subsystem in telemetry Mode 4.

Installation of the computer system and the data display subsystem was completed during this reporting period. Additional elements of the data input subsystem (DIS) were delivered by the contractor and are being installed and checked out. These include the events and status module, and the counters module, which interface with the DIS controller to provide input information to the computer such as the occurrence of spacecraft stimuli and responses, command data, power and signal frequencies, the occurrence of undesired transients, etc. The analog modules for measurement of spacecraft voltages and currents are in the early design phase by the contractor (Astrodata, Inc.) and are scheduled for delivery in December 1966.

Additional functional requirements were established for the STCDS, consisting of the real-time acquisition, processing, recording, and display of the science data automation subsystem outputs to the data encoder and tape recorder subsystems. These data will be recorded on a tape, separate from the STCDS log tape, in a format that enables processing of the data by the Experimenters Reformatting Program (XRPT) in an IBM 7094 system, or other off-line processors. The DAS data and science telemetry data will be displayed in a special format on data display line printer 2, located in the computer area. This printer is also used for the display of data from the system test of a second spacecraft, for program development and debugging, and for nonreal-time test data processing which may be conducted concurrently with a spacecraft system test. These new DAS data processing functions, together with the requirement for processing two spacecraft telemetry data streams simultaneously, necessitated the procurement of four additional data input modules for the DIS; they are scheduled for delivery in early November.

C. Guidance and Control

1. Central Computer and Sequencer Subsystem

Following NASA approval for the *Mariner* Venus 67 Project, preparations for modification and test of the two spare CC&S subsystems (residual from the *Mariner* Mars 1964 Project) were undertaken. The first 3 mo of 1966 were spent mainly in setting up the rework and test area, and testing and repairing the prototype subsystem to be used as a breadboard for the required modifications to the two CC&S flight subsystems. Also, the subassembly test sets were reactivated in preparation for any tests necessary at the subassembly level.

Modifying the *Mariner* Mars 1964 engineering prototype (EP) and type-approval (TA) subsystems to the *Mariner* Venus 67 configuration and testing the completed rework, including compatibility with the operational support equipment (OSE) was accomplished in April. The CC&S modification for the *Mariner* Venus 67 was minimal, involving only the changing of the master time matrix in the end counter (5A3) subassembly. The drawing modifications and changes to the test procedures and specifications required for the project were started. The EP and TA subsystems were used to check out the written test procedures prior to submittal to the specification writers, and prior to their use with flight equipment.

May and June were devoted to modifying and testing the two flight subsystems that had previously been stored at the Spacecraft Assembly Facility (SAF). During post-

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modification tests on the Serial No. MC-04 subsystem, a marginal relay was discovered and replaced in the address register (5A6) subassembly. The test procedures and specifications were finalized and released.

In July the preenvironmental tests, and the preparation for and running of environmental tests, on the MC-04 (M67-1) subsystem were performed.

August was spent troubleshooting, and repairing two broken magnetic core wires found in the maneuver clock (5A4) subassembly in the MC-04 (M67-1) subsystem subsequent to environmental test. The unit was repaired, repeat of environmental test was not considered necessary, and the subsystem was delivered, on time, to SAF September 1.

The Serial No. MC-05 (M67-2) subsystem finished preenvironmental test, was subjected to environmental test, and subsequently postenvironmental tests, with no problems. The time base oscillator was calibrated, and the subsystem was to be delivered to SAF December 7.

In addition to the flight equipment efforts, work was begun on the digital-to-analog (D/A) converter for *Mariner* Venus 67 flight analysis support, similar to that used on the *Mariner* Mars 1964 Project. A major redesign of the *Mariner* Mars 1964 D/A equipment was necessitated by the change in the SFOF data format–both high speed and teletype. The design effort has continued; parts have been ordered and some received, and the fabrication of the first unit was initiated during this reporting period. The first D/A converter is scheduled for delivery the first week in January 1967.

2. Attitude Control Subsystem

In keeping with the objective of retaining the *Mariner IV* configuration insofar as possible, a minimum number of changes to the attitude control subsystem have been made for the *Mariner* Venus 67 spacecraft. These modifications are summarized, by unit, below.

a. Electronics. Two zener reference diodes have been added to each of the +16 and −16 v dc Sun sensor excitation busses to reduce the excitation voltage to +12 and −12 v dc. This lower excitation voltage is required to reduce heat dissipation due to resistance losses, since the heat dissipation due to radiation will increase with proximity to the Sun. In addition, the Earth sensor and roll rate gyro telemetry outputs were previously time-shared; the opening of four connections and the jumpering of two of these connections removes the Earth sensor from this telemetry channel; it is now telemetered on a single channel.

b. Gyro control assembly. There were no modifications to the *Mariner IV* assembly.

c. Jet vane actuators. DC-11 was substituted for Apeizone-N as the shaft-seal lubricant; it has superior sealing qualities at depressed temperatures and may help alleviate the "storage" problem of the lubricants' "drying out" after long periods with no shaft movement.

d. Gas system. The gas valves were reworked for the higher temperatures expected on a Venus mission by reducing the valve travel, decreasing the spring return rate, removing a fillet from the return spring guide pin to permit proper seating of the return pin, adding a tantalum shim on the solenoid air gap to reduce residual magnetism levels, and relapping the valve seat to reduce gas leakage. The Tedlar wrapping around the cabling leading to the valves was removed near the ends of the solar panels to prevent overheating of the cabling. The solar vanes and associated cabling have been removed and the two half-gas systems consequently have become interchangeable with a slight modification of the lower ring harness. With these modifications, the gas system is qualified to operate in the temperature range of −45 to +215°F.

e. Earth sensor. The *Mariner IV* Earth detector has been relocated on the spacecraft bus and the baffling modified to restrict the field of view so that it may be used as a crude roll reference (in the event of failure of the Canopus sensor) for a period of up to 20 days flight from launch. It has been assigned its own telemetry channel, and telemetered indications of Earth's position in the field of view will be used to command spacecraft turns (of a magnitude of ±2.27 deg) to fix the spacecraft position prior to midcourse maneuver.

f. Canopus sensor. With the spacecraft traveling toward the Sun instead of away from it, the location and cone angles of the Canopus sensor on the bus were redefined. The Sun shutter was moved inside the baffling and the baffling redesigned to preclude a Venus stray-light problem. Some suspect parts were removed from the electronics, and the 8× Canopus intensity gate was removed to prevent transient loss of Canopus track due to small particles entering the field of view. The Sun shutter solenoid was redesigned to reduce magnetic interference.

g. Terminator sensor. The narrow-angle Mars gate electronic circuitry has been modified to provide an output to the pyrotechnic subsystem when the terminator of Venus enters the sensor's field of view; this output triggers the mechanism which changes the high-gain antenna pointing angle. The position on the spacecraft bus has been redefined with this objective in mind.

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h. Sun sensors. The primary Sun sensor assemblies have been moved to the sunlit side of the bus and the secondary assemblies to the shadowed side; some changes in cabling were necessary. In addition, the cruise excitation voltages were reduced to compensate for the increased temperatures expected. Null offsets due to reflected sunlight have been reduced by painting portions of the pedestal assembly black, and a temperature transducer has been added to one of the primary assemblies.

i. Planet sensor. A new device, the planet sensor, has been designed and fabricated. It will initiate the science experiments upon sensing the limb of Venus, approximately 1 hr before encounter. The planet sensor is based on the narrow-angle Mars gate with an (approximately) oblong field of view and new electronic circuitry.

3. Thrust Vector Control Assembly

The thrust vector control assembly for *Mariner* Venus 67 consists of four jet vane actuators mounted on a ring, as shown in Fig. 1. The ring holds the vanes in the exit plane of the thrust vector control motor. Autopilot control forces for the spacecraft are provided in all three axes, roll, pitch, and yaw by the vanes deflecting the exhaust gases from the motor during the spacecraft midcourse correction maneuver.

The jet vane actuators for *Mariner* Venus 67 were manufactured and tested in 1964 for use on *Mariner* Mars 1964. They were taken from SAF storage to be reevaluated for flight use on *Mariner* Venus 67 with the following results:

(1) All actuators failed the initial bench tests because of high stiction.

(2) The actuators were reworked and put through flight-acceptance testing. All actuators failed to operate at 32°F, due to shaft binding.

(3) The actuators were reworked a second time and put through flight-acceptance testing again with satisfactory results.

(4) Thrust vector control assembly C-108 was delivered to SAF August 30, 1966.

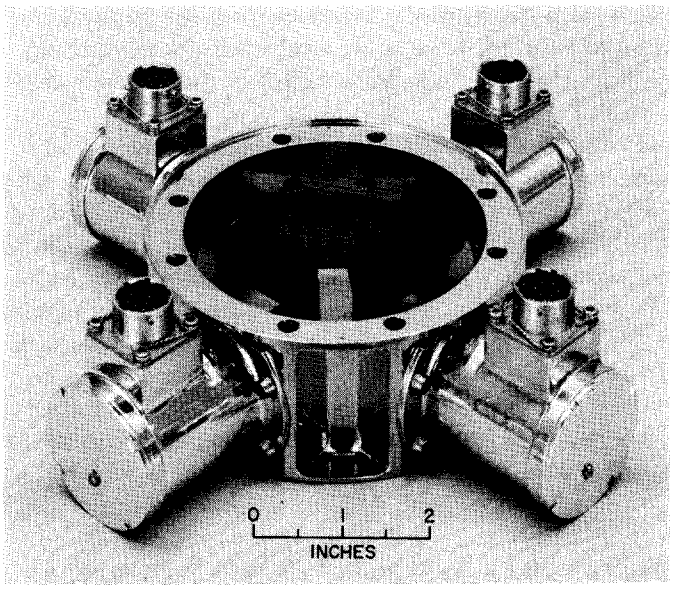

Fig. 1. Thrust vector control assembly

The jet vane actuators were designed and manufactured by the Kearfott Company to perform as required by JPL specification. Twelve usable jet vane actuators were left from *Mariner* Mars 1964. They are mounted on three servo rings, Serial Nos. NF-4, C-104, and C-108. Thrust vector control assemblies S/N C-104 and S/N C-108 were flight-qualified and submitted to SAF for *Mariner* Mars 1964; S/N NF-4 was put through 1 mo of temperature vacuum testing at the flight-acceptance level. The serial numbers of the 12 actuators on the 3 rings are:

(1) Ring NF-4: S/N 2, S/N 4, S/N 7, S/N 32.

(2) Ring C-104: S/N 9, S/N 10, S/N 16, S/N 35.

(3) Ring C-108: S/N 14, S/N 29, S/N 31, S/N 33.

Thrust vector control assemblies S/N C-104 and S/N C-108 were returned to the actuator laboratory from SAF to be reevaluated for *Mariner* Venus 67 use. All actuators on both rings failed in the initial bench tests. They were found to be out of tolerance on stiction. The specification allows a maximum of 0.6 oz-in. stiction. All other performance parameters were within the specified requirements.

A decision was made to rework the actuators by renewing and relubricating the shaft seals. This work was performed on actuators S/N 29, S/N 14, S/N 33, and S/N 31 mounted on ring assembly S/N C-108. Thrust vector control assembly C-108 was then put through flight-acceptance tests in accordance with JPL specification. The operation of all actuators was unsatisfactory at +32°F. Investigation and analysis indicated the shaft seal lubricant used, Apiezon N, to be hardening at +32°F, sufficient to cause marginal or unsatisfactory performance.

A qualitative comparison of three seal lubricants, Shell APL, Apiezon N, and DC-11, through the temperature range from +20 to +200°F was made. The comparison

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tests were made using an actuator shaft and shaft seal O-rings. Apiezon N became very hard at +32°F, Shell APL very stiff. DC-11 indicated a slight, nearly imperceptible increase in viscosity from +200 to +20°F. The JPL chemistry laboratory was consulted, in an effort to obtain some quantitative data on these lubricants. The tests were run in a vacuum of 10^{-6} torr at a temperature of 138°C. The following data were reported with regard to weight loss under conditions of vacuum and temperature.

(1) Apiezon had a weight loss of 0.1% in 38 hr.

(2) Shell APL had a weight loss of 7.99% in 38 hr.

(3) DC-11 had a weight loss of 1.79% in 24 hr.

Weight loss in Shell APL causes an increase in viscosity because of losing its lighter fractions. This is not true for Apiezon N and DC-11; they do not experience a physical change as a function of weight loss. As a result of these tests and data, a decision was made to change the shaft seal lubricant in all *Mariner* Venus 67 actuators from Apiezon N to DC-11.

The actuators on ring C-108 were reworked, incorporating the changed lubricant. During the preflight-acceptance bench tests actuator S/N 29 was observed to have a gross leak at the shaft seal. Actuator S/N 4 was reworked and mounted on ring C-108 to replace S/N 29. Ring C-108 with actuators S/N 4, S/N 14, S/N 31, and S/N 33 was then put through the flight-acceptance test procedures. After the temperature vacuum test, actuator S/N 4 was observed to have reduced torque gain. The specification requires 0.0336 ±0.005 oz-in./ma; S/N 4 had dropped to 0.021 oz-in./ma.

The seal in actuator S/N 29 was reworked. Actuators S/N 29 and S/N 4 were remagnetized, carefully stabilized, and then put through flight-acceptance tests. Actuator S/N 4 again lost torque in the temperature vacuum environment.

Actuator S/N 29 was mounted on ring C-108 to replace S/N 4, which was rejected for flight use; actuator S/N 4 has had a history of being difficult to magnetize and of losing its torque.

Thrust vector control assembly C-108 with actuators S/N 14, S/N 29, S/N 31, and S/N 33 was qualified for *Mariner* Venus 67 flight use and delivered to SAF August 30, 1966.

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II. *Mariner* Mars 1969 Project

A. Introduction

The *Mariner* Mars 1969 Project was initiated in late-December 1965 and formally tasked on February 1, 1966. The primary objective is to conduct two Mars flyby missions in 1969 to make exploratory investigations of the planet which will set the basis for future experiments—particularly those relevant to the search for extraterrestrial life. The secondary objective is to develop the technology needed for succeeding Mars missions.

The spacecraft design concept will be based on the configuration of the successful *Mariner IV* spacecraft, with considerable modifications to meet the 1969 mission requirements and to enhance mission reliability.

The launch vehicle will be the *Atlas/Centaur* SLV-3C. This vehicle, developed under contract for and direction by the Lewis Research Center by General Dynamics/Convair, has a single- or double-burn capability in its second stage and a considerably increased performance rating over the *Atlas D/Agena D* used in the *Mariner IV* mission.

Mariner Mars 1969 missions will be supported by the Eastern Test Range launch facilities at Cape Kennedy, the tracking and data acquisition facilities of the Deep Space Network, and other NASA facilities.

Six planetary-science experiments have been tentatively selected by NASA (subject to integration capability) for the *Mariner* Mars 1969 missions; they are listed in Table 1. Additionally, a planetary-approach-guidance engineering experiment will be incorporated to test the feasibility and flight-readiness of on-board optical sensors and associated data processing techniques necessary for optical approach guidance.

Table 1. *Mariner* Mars 1969 scientific investigations

Experiment	Scientific investigator	Affiliation
Television	R. B. Leighton[a]	California Institute of Technology (CIT)
	B. C. Murray	CIT
	R. P. Sharp	CIT
	N. H. Horowitz	CIT
	J. D. Allen	Jet Propulsion Laboratory (JPL)
	A. G. Herriman	JPL
	L. R. Malling	JPL
	R. K. Sloan	JPL
	N. Davies	Rand Corporation
	C. Leovy	Rand Corporation
Infrared spectrometer	G. C. Pimentel[a]	University of California, Berkeley (UCB)
	K. C. Herr	UCB
Ultraviolet airglow spectrometer	C. A. Barth[a]	University of Colorado
	W. G. Fastie	Johns Hopkins University
Infrared radiometer	G. Neugebauer[a]	CIT
	G. Munch	CIT
	S. C. Chase	Santa Barbara Research Center
S-band occultation	A. J. Kliore[a]	JPL
	D. L. Cain	JPL
	G. S. Levy	JPL
Celestial mechanics	J. D. Anderson[a]	JPL

[a]Principal investigator.

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During the reporting period, mission design and system design of the spacecraft continued; the interaction and exploration of scientific experiment requirements affected both efforts. A quarterly review of the Project was conducted September 15.

Scientific instrument development efforts were in the process of being procured by contract. Litton Industries, Inc., was selected as contractor for the data automation subsystem on November 2.

Contractor efforts for the four telecommunications subsystems are under way; the RF subsystem contract was signed in the previous report period, and telemetry, command, and data storage contracts were signed in September. In guidance and control, the attitude control contract was initiated last period, CC&S in September; TRW, Inc., was selected as contractor for the power subsystem in October, and negotiation is under way. Electro-Optical Systems, Inc., was selected last report period for the approach guidance, and the contract is being prepared. The engineering mechanics effort has been under way for some time, as reported in SPS 37-41, Vol. II and Section D of this volume. Selection of contractors and initiation of development for the pyrotechnic and propulsion subsystems are expected in the next period; initial description of these subsystems is given in Section G of this SPS.

B. Project Engineering

1. Spacecraft System Design

a. Flight profile. The *Mariner* Mars 1969 flight profile may be divided into the phases of launch, cruise, maneuver, encounter, and playback. The mission objectives and technical approach impose requirements on each of these phases that reflect, on the design of the spacecraft, requirements that result in changes from the *Mariner* Mars 1964 configuration.

In the *Mariner* Mars 1969 design, as on all *Rangers* and *Mariners*, the sequencing through the mission phases is primarily controlled by onboard logic, the central computer and sequencer (CC&S). The *Mariner* Mars 1964 CC&S cannot be constructed to 1966 standards of quality assurance. Thus it is necessary to develop a new design. The design selected is a programmable sequencer, with a 128-word core memory that can be updated or modified by ground command. This programmable sequencer allows a versatility in mission operations heretofore unobtainable in *Ranger* and *Mariner* missions.

Launch phase. In the launch phase, the *Mariner* Mars 1969 spacecraft will be boosted into the interplanetary transfer trajectory to Mars by the *Atlas/Centaur* launch vehicle. Substitution of the *Centaur* for the *Agena* stage, of course, imposes new requirements on the spacecraft design, and subjects the spacecraft to a new acoustical and vibrational environment. The launch profile may be either parking orbit or direct ascent. The shroud will be the *Surveyor* design and is RF transparent. As two traveling wave tube power amplifiers will be incorporated in the radio subsystem, for which there is no low power mode, the present flight profile assumes that traveling wave tube power is on, and radiating 10 w of RF from lift-off.

At the termination of the launch phase, the spacecraft is separated from the *Centaur* launch vehicle, on command from the *Centaur*. The use of *Centaur* requires a new mechanical and electrical interface. The adapter uses the *Mariner* Mars 1964 design at the separation plane, and the *Surveyor* design at the field joint, with JPL maintaining design responsibility down to the field joint. The use of the *Surveyor* shroud requires that the spacecraft umbilical lines be routed down through the adapter and out the *Centaur* umbilical island.

Following separation from the launch vehicle, the spacecraft acquires the Sun, and immediately commences Canopus acquisition. No magnetometer roll calibration, as performed on *Mariner* Mars 1964, will be required, for no magnetometer is included in the *Mariner* Mars 1969 science payload. The Canopus tracker will incorporate an adaptive gate for star brightness, which will be successively lowered by cyclic command from the CC&S, until a celestial object, hopefully Canopus, is acquired. Then the spacecraft enters into the cruise phase of the mission.

Cruise phase. The science payload contains no instruments which are operated during the cruise phase of the mission, that is, no interplanetary experiments. Thus, the data mode for the launch and cruise phases consists solely of engineering data and is called "the cruise data mode." Since neither magnetometers nor particle detectors are included, the project requirement is removed for magnetic cleanliness and absence of radioactive materials.

The spacecraft starts the cruise phase transmitting engineering data at 33⅓ bits/sec over the low-gain antenna. During the flight, the cone angle of the Canopus tracker is updated by programmed command by the CC&S. When the communications performance drops to the sum of the negative tolerances, the CC&S commands the telemetry subsystem to lower the bit rate to 8⅓ bits/sec. The communications performance drops again into

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greyout, this time at 8⅓ bits/sec, and after several days in greyout, the CC&S commands the radio subsystem to switch the power amplifier to the fixed high-gain antenna. The communications performance rises rapidly at 8⅓ bits/sec on the high-gain antenna, and after a suitable performance margin has been obtained, the telemetry bit rate is switched to 66⅔ bits/sec, the bit rate for the encounter mode, and for the first part of the playback mode. Using the *Mariner* Mars 1964 radio subsystem essentially, 66⅔ bits/sec will be possible because of the shorter communication distance for the 1969 encounter than for the *Mariner* Mars 1964 encounter.

Ten days prior to arrival at the planet, the approach guidance subsystem will be turned on, this being the last event of the cruise phase of the mission.

The approach guidance subsystem optically tracks the center of Mars during the period from 10 days until 12 hr before encounter. The cone and clock pointing angles to the center of Mars are measured in the spacecraft reference system. These angles, in conjunction with the Sun and Canopus sensor error signals, are telemetered to the ground where they are processed and used in an orbit-determination program to compute the near-planet trajectory of the spacecraft.

The cruise phase will be interrupted early in the flight in order to perform the trajectory correction maneuver. As on *Mariner* Mars 1964, the capability for a second maneuver is provided for in the design. On *Mariner* Mars 1969, redundancy in the maneuver logic will be present, in that the CC&S will have a separate fixed sequencer for the maneuver functions.

Maneuver phase. In the tandem maneuver mode, the CC&S will compare the output of the programmable sequencer against the output from the fixed sequencer. In the event of a discrepancy, the maneuver will be automatically aborted. The capability will exist, by ground command, to select the correctly functioning sequencer, and to repeat the maneuver sequence.

Encounter phase. The encounter phase is divided into the far-encounter sequence, the near-encounter sequence, and occultation.

At the start of the far-encounter sequence, the scan control, television, infrared radiometer, and data automation subsystems are energized. The scan platform, which has degrees of freedom in both cone and clock, supports the two-camera television, ultraviolet and infrared spectrometers, and infrared radiometer. After the science instruments have warmed up, and sufficient time has passed for the Mission Operations System (MOS) to evaluate the performance of the spacecraft in the far-encounter mode, the scan platform will be commanded from the erected near-encounter position with cone angle equal to approximately 120 deg, to the far-encounter position. If the planet brightness is above threshold for the far-encounter scan sensor, the scan control subsystem switches from open-loop pointing to a closed-loop mode and servos on the planet. A series of TV pictures will be taken by the narrow-angle television camera between closest approach minus 48 hr and closest approach minus 12 hr. The picture data will be stored on a *Mariner* Mars 1964-type tape recorder using an analog recording scheme. At the conclusion of the far-encounter sequence, the scan platform will be commanded back to the near-encounter position.

The near-encounter sequence commences several hours prior to closest approach. At that time all the science instruments will be energized. The scan platform is fixed at a cone angle of 120 deg, a position which is identical on both spacecraft and is fixed at one of a number of selectable clock positions. The clock position can be stepped by ground command through a number of discrete positions in 3-deg increments. Thus, no servo sensor or closed-loop planet tracking is presently envisioned for the near-encounter sequence.

About 20 min prior to the start of near-encounter data taking, cooling of the infrared spectrometer detectors will be initiated by CC&S command. When the planet comes within the view of the narrow-angle television camera, near-encounter data taking will commence.

Because of the large increase in the required amount of data storage, due not only to the increased number of TV pictures, but also to the increased number of lines and elements per picture, a new mechanization is proposed for the data storage subsystem. The new mechanization permits a 30-fold increase in data storage, from 5.2×10^6 bits on *Mariner* Mars 1964 to 1.8×10^8 bits for *Mariner* Mars 1969. The mechanization consists of two *Mariner* Mars 1964-type endless loop tape recorders. One recorder stores 2.3×10^7 bits in digital form; the other stores 1.57×10^8 equivalent bits in an analog manner.

During the near-encounter data-taking sequence, the scan platform will be dropped from a cone angle of 120 deg to one of approximately 90 deg, in order to extend the duration of the sequence. The data taking continues

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until both the analog recorder and the digital recorder are full.

Both the near- and far-encounter will be conducted in the encounter data mode at 66⅔ bits/sec. The encounter data mode of *Mariner* Mars 1969 is analogous to the cruise data mode of *Mariner* Mars 1964 in that 140 bits of engineering data will be followed by 280 bits of science data. There will be no data mode equivalent to the all-science Data Mode 3 of *Mariner* Mars 1964.

Approximately 1 min after the data taking has stopped, the spacecraft enters occultation, and will be occulted from Earth for approximately 30 min. After exit from occultation, several hours of engineering data will be required for engineering analysis of the spacecraft by the MOS, prior to the start of the playback phase of the mission.

Playback phase. The playback phase begins at a bit rate of 66⅔ bits/sec, until the communication performance starts to degrade, and then the bit rate is commanded to 33⅓ bits/sec for the remainder of the mission.

The present mission plan calls for playback of the recorded data once, which at 66⅔ bits/sec would require 35 days, assuming 2 hr of engineering telemetry per day.

The *Mariner* Mars 1969 playback phase is more complicated than that of *Mariner* Mars 1964 in that periodically, roughly every 3 days when transmitting at 66⅔ bits/sec, data must be transferred from the analog tape recorder, through an analog-to-digital converter, and rerecorded on the digital machine.

b. Design implementation. The implementation of the technical approach to satisfy the mission objectives calls for major changes to the *Mariner* Mars 1964 spacecraft subsystems.

Structure. The basic structure must be modified to accommodate the changes to other subsystems requirements and the over-all increase in spacecraft weight from the 576-lb *Mariner* Mars 1964 to the 844-lb *Mariner* Mars 1969 (Table 2). In addition, a new adapter section for mating the spacecraft to the launch vehicle is required to accommodate the launch vehicle change from *Agena* to *Centaur*. Included in this change is the relocation of the umbilical connector.

Mechanical devices. The mechanical devices subsystem is newly designated, incorporating a number of functions previously assigned to other subsystems. Specifically, this includes solar panel boost damper, low-gain antenna damper, solar panel deployment and latch mechanisms including the switch assemblies for indications of deployment-initiated timer, spacecraft separation mechanisms, and the spacecraft V-band. Nearly all of these devices require modification for *Mariner* Mars 1969 use. Included in this subsystem is the scan platform, which must be redesigned to incorporate 2 degrees of freedom and accommodate a six-fold increase in weight of instruments mounted on the platform.

Table 2. Weight summary

	Mariner Mars 1964	*Mariner* Mars 1969
Science		
Platform	9.5	57.0
Bus	42.4	69.0
Total	51.9	126.0
Data Storage	17.1	36.0
Subtotal	69.0	162.0
Engineering	506.6	574.5
Subtotal	575.6	736.5
Adapter		50.0
Contingency		57.6
Total	575.6	844.1

Temperature control. The temperature control subsystem will probably require heater power for the scan platform. A temperature control flux monitor is incorporated in lieu of the *Mariner* Mars 1964 absorptivity standard.

Pyrotechnic. The pyrotechnic subsystem will require modifications to meet new requirements from the other subsystems, such as science instrument cover releases and the infrared spectrometer (IRS) gas bottle valve activations.

Propulsion. No changes are planned for this subsystem.

Power. The power subsystem is completely redesigned; the boost regulators, inverters, battery charger, and power distribution logic will be redesigned. The primary timing reference is moved from the CC&S into the power subsystem. Structural changes require the incorporation of additional Sun sensors. N/P solar cells will be used to provide increased protection against solar panel damage.

Science. The mission objectives call for a complete new science subsystem including television, data automation, infrared radiometer, ultraviolet spectrometer, and infrared spectrometer. The approach guidance function requires a completely new subsystem.

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Radio. The radio subsystem will incorporate two traveling wave tube (TWT) power amplifiers, in place of the one TWT and one cavity amplifier used on *Mariner* Mars 1964. The receiver equipment is moved from Bay 5 to Bay 6 to make room for two tape recorders in Bay 5. To simplify the switching and to eliminate any uplink antenna interference, the receiver will be attached to only the low-gain antenna.

Telemetry. The telemetry channel assignments are changed to accommodate the new requirements of the other subsystems. New data modes are incorporated and three bit rates are required. It is necessary to process digital data from the approach guidance and the CC&S subsystem.

Command. In the command subsystem, the isolated switch outputs are reallocated among the DC commands.

Data storage. Two *Mariner* Mars 1964 tape recorders are included in the *Mariner* Mars 1969 design, with one machine recording data in an analog manner. Both the record and playback electronics are changed and an A/D converter is added to digitize the analog data.

Attitude control. In the attitude control subsystem, the Canopus tracker incorporates new star tracking logic, including an adaptive gate. The Earth detector is deleted and all attitude control electronics are repackaged.

Scan control. The scan control subsystem is totally new. Control is required for the 2 degrees of freedom scan platform. In the far-encounter mode, a closed-loop servo is required. For the near-encounter, only open-loop pointing is used. The scan control subsystem also provides open-loop pointing for the ultraviolet spectrometer.

CC&S. The CC&S subsystem incorporates a memory-oriented programmable sequencer, and has redundant capability for maneuver sequencing.

2. System Analysis of Primary Power Capability

One of the major constraints on the system of the *Mariner* Mars 1969 spacecraft is power. This is brought about by a substantial increase in the power required over that of *Mariner* Mars 1964 for the various subsystems in operation at encounter. It is anticipated that 1969 will be a year of intense solar flare activity compared to 1964–65, at the time of the *Mariner IV* flight. This is important to the power picture because proton bombardment resulting from solar flares produces power degradation of the solar cells, limiting the maximum power that can be drawn from them.

Progress has been made in obtaining higher efficiencies from the power conversion equipment for the *Mariner* Mars 1969 spacecraft. In addition, it has been decided to discard the *Mariner* Mars 1964 solar panel design in favor of improved designs. This means that the two spare *Mariner* Mars 1964 panels can no longer be used since they will not be compatible. It also means that N/P solar cells, which are much more resistant to the effects of solar flares than their P/N counterparts on *Mariner* Mars 1964, can be used. It has been decided to study ways and means of increasing the active solar panel area since it has been demonstrated that a small increase in area, with its resulting increase in power-producing capability, would result in a greatly increased probability of being able to perform the encounter without the necessity of sharing power between the battery and the solar panels.

The probability of being able to supply a given amount of power from the solar panels at encounter as a function of the power is shown in Fig. 1. It can be seen that if the total demand on the panels is under 200 w, the probability of being able to supply the entire power from the panels alone is approximately 100%. It can also be seen that at the other extreme, if the demand is more than 332 w, the capacity of the panels has been exceeded, even

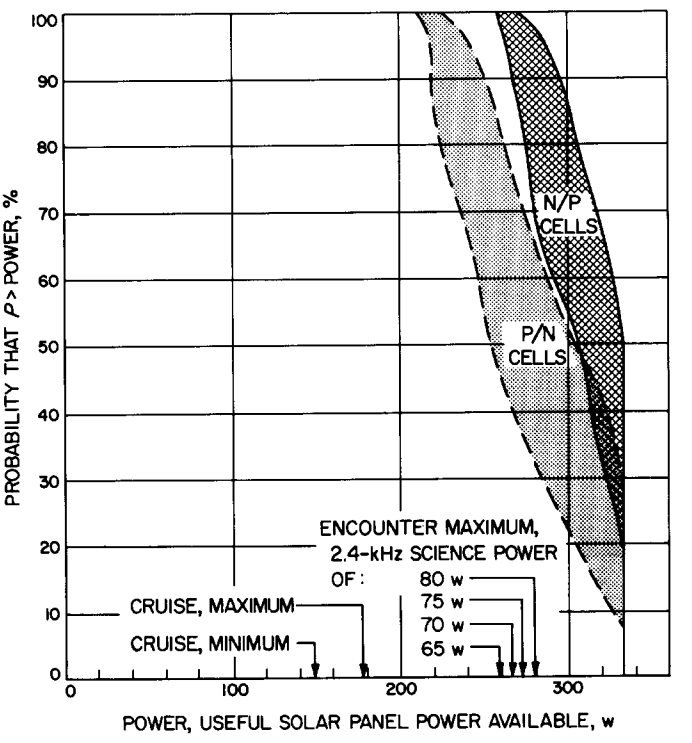

Fig. 1. Probability of supplying power from solar panels at encounter

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without any degradation at all, and sharing would be required. Between these two extremes is a range of probabilities that depends on how much power is demanded.

The N/P cells, represented by the cross-hatched area in Fig. 1, can supply more power than the P/N cells under the same conditions that cause the degradation. Undegraded, they are about equivalent in their power-producing capacity, as represented by the straight vertical grid lines. As the power demand by the spacecraft increases, the probability of being able to supply this power goes down. Assuming a 2.4-kHz science power load of 80 w (which is representative of the current spacecraft design), the encounter power demand from the panels for the entire spacecraft, including science, is 280 w. Provided that gyros are not on at this time, the P/N cells give at *best* only a 65% probability of being able to perform the encounter without battery sharing. Switching to N/P cells gives in the worst case a 68%, and in the best case a 97%, probability of being able to supply the required power without sharing.

A solar flare model was developed by the JPL Environmental Requirements Section, representing its best estimates of what can be expected in the way of solar proton activity during a representative 6-mo journey to Mars in 1969. It is presented as a curve of "six-month probability of flux occurrence greater than N for energies above 5 Mev versus N, the integral flux." The analysis is restricted to protons with energies above 5 Mev since this represents the cutoff level of the protective cover glass on the solar cells.

By folding (eliminating the common flux parameter) the curve of the "percent degradation of the panels that occur for various integrated flux levels" and the curve of "six-month probability of flux occurrence greater than N for energies above 5 Mev versus N, the integral flux," a curve of the probability of degradation of the panels is obtained. In order to relate the percent degradation to available power, it is necessary to know the undegraded power available at encounter. For 70 ft² of active panel area, and considering such things as manufacturing tolerances, test and measurement tolerances, and the fact that the useful power available is 5% less than the peak of the power curve, it turns out that for the early-August 1969 arrival dates presently under consideration, 332 w is the magic number of useful watts available from the solar panels. The percent degradation for each probability point is then applied to 332 w and we get this curve with degradation starting at 332 and going down from there.

The curves are quite steep and by shifting all of the curves to the right just a small amount, degradation becomes an almost nonexistent problem. One way of doing this is by increasing the undegraded power-producing capacity of the solar panels by simply increasing the active panel area or, effectively, adding more cells.

Table 3 shows the results of a tradeoff study to consider the effects of reducing the science power load and increasing the active panel area. The present science load is 80 w and the P/N cell-type panel with 70 ft² of area is included here for comparison. There is a great improvement evident by just switching to N/P cells with the same area, as discussed before, and here it can be seen that there is even greater improvement in the situation just by adding a few square feet of panel area. Even with the 80-w science load, 75 ft² of N/P cells bring us up to the 95-to-100% probability region, and 82 ft² eliminates the problem altogether.

Table 3. Probability that $P >$ power, %

Panel area	70 ft²				75 ft²		82 ft²	
Undegraded power	332 w				356 w		389 w	
Cell type	P/N		N/P		N/P		N/P	
2.4-kHz science power	High estimate	Low estimate	High estimate	Low estimate	High estimate	Low estimate	High estimate	Low estimate
80 w	65	32	97	68	100	95	100	100
75 w	70	37	99	82	100	100	100	100
70 w	77	41	100	90	100	100	100	100
65 w	83	47	100	97	100	100	100	100

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Another problem is the 2.4-kHz main inverter. The present new design is sized for 155 w and this limit cannot be exceeded without extensive redesign. The present spacecraft loads on this inverter are right up to the limit and studies will be conducted to see if the load level can be reduced, at least slightly.

C. Systems

1. Trajectory Design

The preferred arrival date period for the *Mariner* Mars 1969 missions is August 1–10, 1969. Two fixed arrival dates will be utilized for the two spacecraft and the dates finally selected must be separated by a minimum of 5 and a maximum of 9 days in order to satisfy encounter mission operations requirements.

The decision was made in October to use only the direct-ascent mode for the missions. As a result of the large negative departure asymptote declinations for the 1969 opportunity, however, a launch period of only 30–35 days can be achieved by flying planar ascent trajectories and assuming a maximum southeast launch azimuth of 115 deg (east of north). By using either more southerly launch azimuths or by employing a southerly "dog-leg" maneuver from the 115-deg launch azimuth, it is possible to open the launch period 15–20 days earlier, thus increasing the total launch period available to 45–55 days. As a result of AFETR safety constraints, the dog-leg option is much more probable than the extension of initial launch azimuth south of 115 deg. The earliest possible opening of the launch period is February 16, 1969, with the close of the launch period governed by excessive injection energy requirements around April 15, 1969.

Fig. 2 defines the approach aiming plane at Mars. It can be seen that various flyby trajectory constraints can be expressed in terms of aiming-point contours in the **R**, **T** plane. Fig. 3 illustrates a typical aiming diagram (to scale) for one of the *Mariner* Mars 1969 arrivals. The shaded area represents the broad region of aiming-point acceptability, while the smaller interior "windows" represent the preferred aiming zones. The southerly zone is based upon the scientific desires to view the thaw and ice regions near the edge of the southern polar cap, while the more northerly aiming zone is based upon the desire to view the large number of light and dark surface feature variations which are contained within the Martian latitude band from about 30 deg south to 10 deg north. For a nominal flyby periapsis altitude of 3000 km, picture resolution will be about 100 m per TV line for the high-resolution camera system.

2. Planetary Quarantine

It is essential that precautions be taken to ensure a high probability that Mars will not be contaminated with terrestrial organisms prior to the successful completion of an adequate spectrum of biological experiments. The need to perform a large number of experiments will require that several missions be flown to Mars, and the probability of Mars contamination by any single mission must therefore be held to a very small number. For each launch of the *Mariner* Mars 1969 mission, the currently specified probability that Mars will be contaminated shall not exceed 3×10^{-5}. It is, of course, essential to define what is meant by the term "contaminated." It is possible to impact Mars with either large organism populations (unsterilized *Centaur* stage or *Mariner* spacecraft) or very small populations (various effluxes such as attitude control gas, spalling, outgassing, propulsion system exhaust, and micrometeroid ejecta). In regard to the former category, contamination shall be "defined" to have occurred if impact occurs, even though it is generally felt that the Martian environment is sufficiently hostile to terrestrial organisms (even in large populations) as to result in a low probability of survival and replication. In regard to the second category, concerning single or very small organism populations, impact shall not be defined as tantamount to contamination. In fact, the probability of contamination by small particles containing only a few organisms shall depend upon the product of a number of such probabilities as:

(1) Being viable at the time of release from the spacecraft.

(2) Surviving the ejection process.

(3) Achieving an impact course in the presence of solar radiation pressure.

(4) Surviving the ultraviolet radiation during cruise to the planet.

(5) Surviving atmospheric entry heating.

(6) Being released into the Martian environment.

(7) Surviving and multiplying in the Martian environment.

In order to ensure a sufficiently low probability that either the *Centaur* stage or *Mariner* spacecraft will impact Mars, it is often necessary to bias the trajectory aiming point a safe distance from the planet. The amount of biasing that may be required depends upon the size of the predicted trajectory dispersions, as well as the reliability of subsequent maneuvers or trajectory alterations. If subsequent (unused) maneuvers had unit reliability, no trajectory biasing would be required for the

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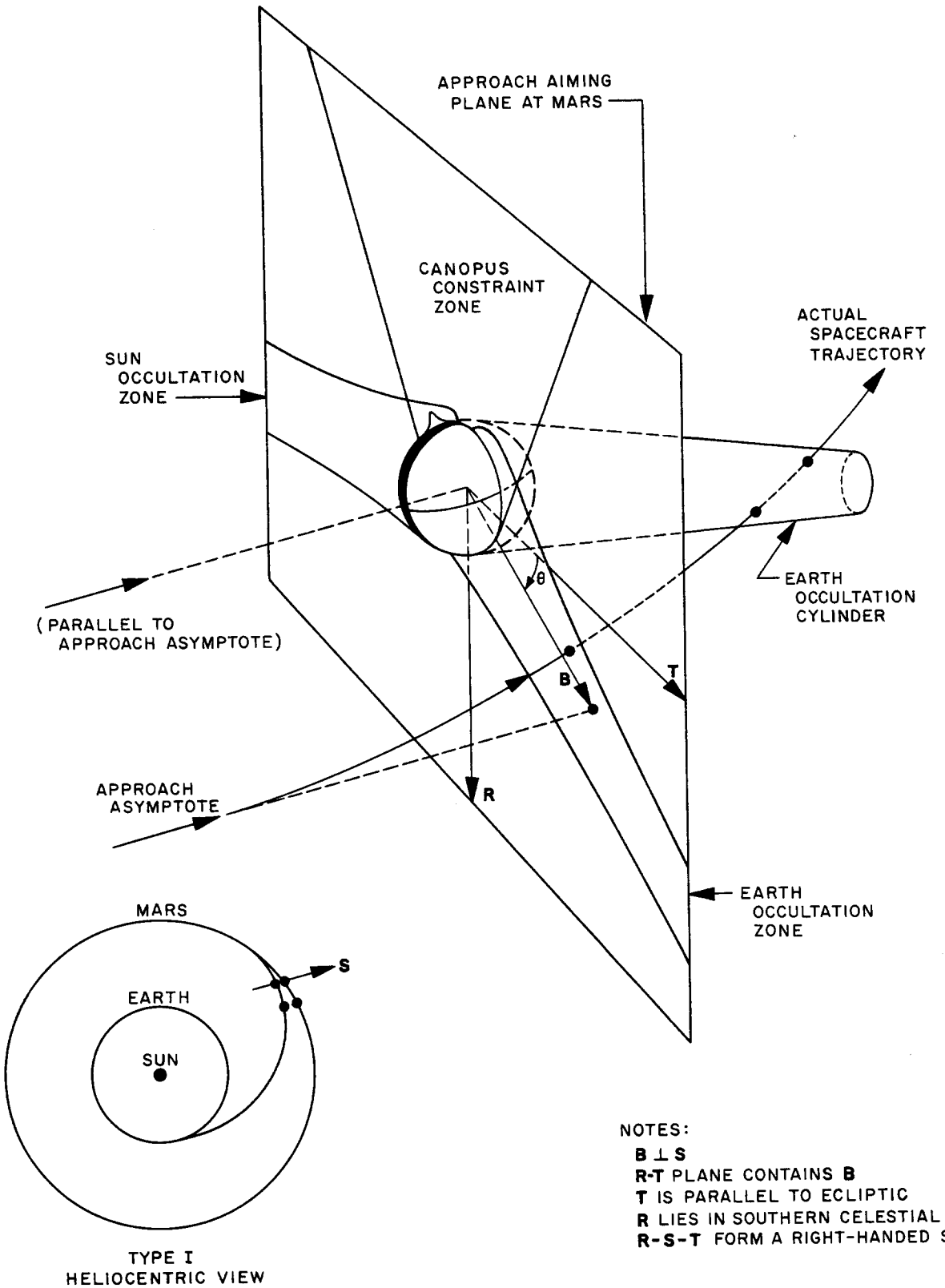

Fig. 2. Definition of approach aiming plane at Mars

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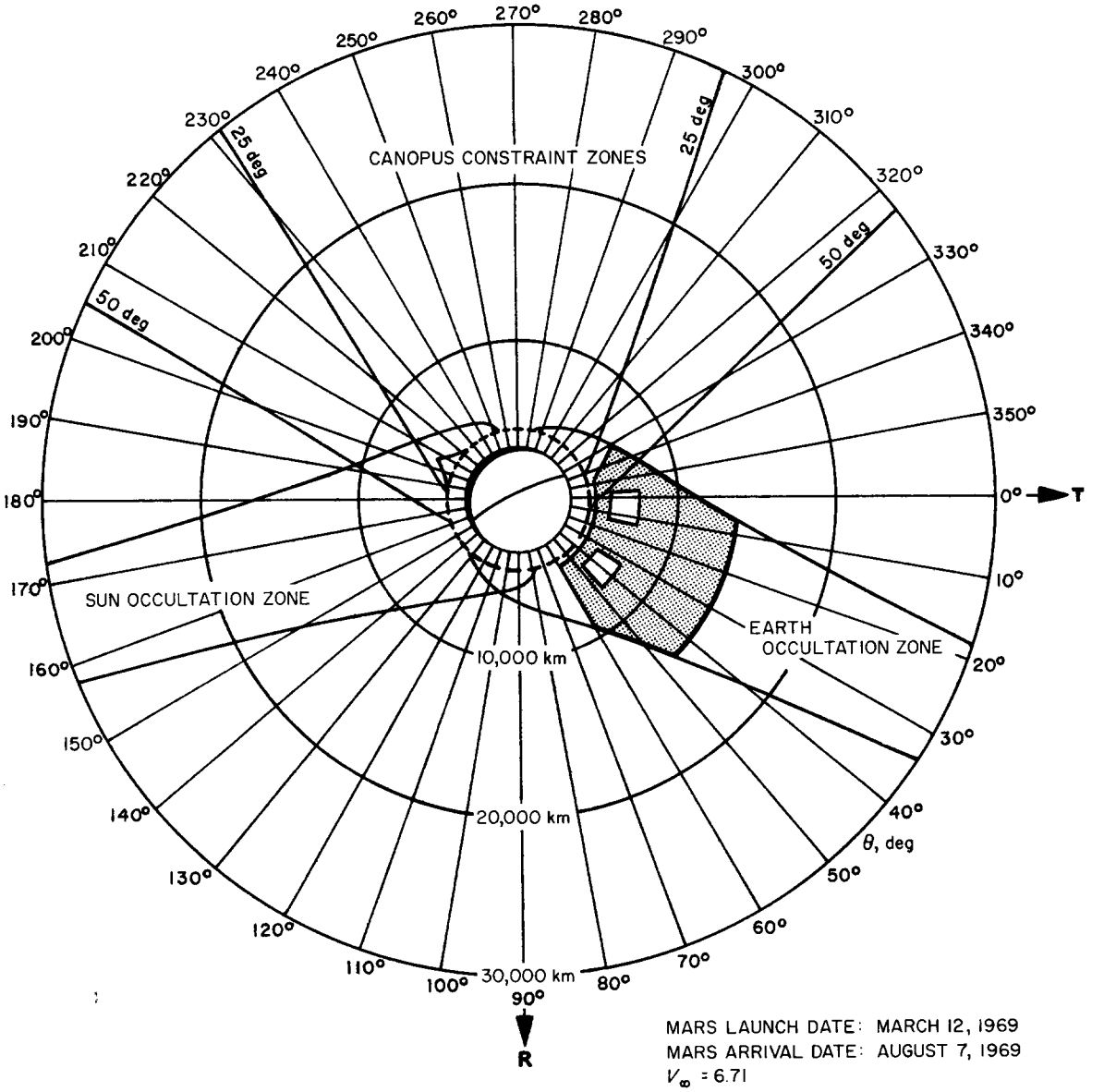

Fig. 3. Typical approach aiming diagram for 1969 Mars opportunity

earlier maneuvers. As this is not the case, however, specific maneuver reliability calculations must be made to determine whether aiming point biasing will be necessary. For the 1969 Mars mission, these calculations indicate that biasing will definitely be required for the injection aiming point and, in some cases (for the more sensitive trajectories having February launch dates), for the first midcourse maneuver aiming point as well.

A study is currently in process to determine the dependency of mission value upon various probability-of-contamination constraint levels. For different assumed launch vehicle and spacecraft maneuver reliabilities, information is currently being generated to show the optimum risk allocation between the injection and first maneuver aiming points, the closest aiming point locus for one midcourse maneuver, the probability of passing through various flyby zones after one midcourse maneuver, etc. This information will be presented to NASA during the next report period, for the purpose of providing the necessary mission tradeoff considerations before the final planetary quarantine constraint is specified precisely.

3. Orbit Determination

In the course of designing the *Mariner* Mars 1969 spacecraft, the question has arisen concerning whether

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to point the science platform by ground command (based on radio-tracking orbit estimation) or by on-board closed-loop tracking of the planet. To resolve this question, it is necessary to determine the platform pointing-accuracy requirements, orbit determination (OD) capability as a function of time during the flight, latest time permissible for updating the platform position, and the pointing-accuracy capability of an on-board closed-loop system.

Currently, the tightest pointing-accuracy requirement formally requested is that of the ultraviolet spectrometer (UVS). Information regarding the neutral molecular gases (N_2, O_2, CO, CO_2) will be degraded if the long axis of the UVS field-of-view slit is not tangential to the bright planet limb at the time of limb intercept (actually at approximately 100 km above the surface at the limb). The UVS experimenter has expressed a desire that the alignment error be less than 3 deg with a probability of 0.9. This would correspond to a ± 1.65-σ one-dimensional gaussian error, and hence a 3-σ slit misalignment error of ± 5.45 deg. Even with perfect OD knowledge, the spacecraft pointing capability can only control the slit misalignment to about ± 3 deg, 3 σ. This would then allow the OD to contribute about $[(5.45)^2 - (3.0)^2]^{1/2} = 4.55$ deg, 3 σ. This in turn corresponds to an error of ± 250 km, 3 σ, in a direction normal to the impact parameter **B** and in the **R**, **T** plane (refer to Fig. 2). It is important to point out that the 3-deg slit-alignment requirement constitutes only about ⅓ of the total value of the UVS experiment.

Studies are currently in process to determine whether the 250-km OD accuracy requirement can be met with confidence. Initial results indicate that such an accuracy level could be reliably met by using the JPL Double Precision Orbit Determination Program and processing radio tracking data obtained as late as several hours prior to closest approach. A lower confidence level exists in meeting the above accuracy requirement with the JPL Single Precision Orbit Determination Program.

As an alternative, on-board planet sensing could be used to provide closed-loop pointing of the science platform. It has been estimated that such a technique could achieve a UVS slit-alignment accuracy of about 3 to 4 deg, 3 σ, which would certainly be satisfactory. It is extremely important to resolve this closed-loop versus radio-tracking question of platform control as soon as possible. Further progress will be reported in a subsequent SPS.

D. Engineering Mechanics

1. Introduction

The Engineering Mechanics effort for *Mariner* Mars 1969 includes responsibility for four subsystems (structure, cabling, temperature control, and mechanical devices) and a number of broad supporting functions (mechanical systems, applied mechanics, parts, materials, and packaging). Mechanical systems and packaging, together with elements of other functions, are part of the spacecraft mechanical design. The applied mechanics function includes structural and dynamic analysis and testing, with particular attention to launch-phase dynamics.

Subsystem and configuration efforts were described in SPS 37-41, Vols. II and VI. The remaining supporting engineering mechanics functions are discussed in this report.

2. Materials Engineering

It is the responsibility of the materials engineering discipline to provide expert help in the use of materials and associated processes in support of project activities. This is achieved through the preparation of controlling documents and direct technical contact with the engineering design, manufacturing, quality assurance, and reliability activities.

Although materials engineering at JPL is a service function supplied by the Engineering Mechanics Division, spacecraft subsystem contractor personnel provide the direct day-to-day materials and processing support to their respective subsystems, working closely with JPL materials engineering to assure consistency in the use of materials, processes, and control specifications.

Preliminary and detail design of each spacecraft subsystem is done primarily by contractors working under JPL technical direction. This is a considerable change from past projects when the majority of design was done in-house. This has changed the materials and processes support role from that of direct contact with designers to an indirect, technical overseeing of the contractor's material and process activities. However, because the *Mariner* Mars 1969 spacecraft is based on the *Mariner* Mars 1964 design, a considerable amount of technical interchange must take place to convey *Mariner* 1964 information. This results in the establishment and maintenance of a very close contact between JPL materials personnel and contractor counterparts.

Because the majority of hardware is being manufactured by subsystem contractors, the material and process support for the manufacturing phase of the project will come principally from the contractor's own personnel. During the manufacturing phase, contractor's material and process activity will be under JPL materials and

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process surveillance. Assistance from JPL will be provided if necessary to solve any serious technical problems.

Quality assurance is supported in several areas, most important of which is the interpretation of nondestructive test results, i.e., radiography, ultrasonic and penetrant. Another area where support is provided is the interpretation of mechanical property and chemical test results on materials, especially when a deviation from specification requirements exists. Lastly, failures resulting from material defects or improper processing are analyzed to determine the proper corrective action to be taken.

The reliability activity is supported by establishing material and process control lists for each spacecraft subsystem. Each subsystem contractor has the choice of using materials and processes selected from the JPL preferred lists or selecting his own and having them approved by JPL.

3. Applied Mechanics

The structures and dynamics effort provides confidence in the structural integrity of the spacecraft and in the efficient utilization of structural weight within project constraints. This includes definition of structurally significant spacecraft loads and environments, determination of structural element loads from composite spacecraft dynamic analyses, performance of detailed stress analyses, active participation in structural design, and direction of structural and dynamic test programs. In general, this effort is concentrated on structural elements which are primarily affected by static loads and by vibration inputs with frequencies below approximately 200 Hz. Consultation on loads, environments, and structural design is provided when required for packages, instruments, etc., not usually considered to be of primary structural significance. Additional activities relate to separation dynamics, to consideration of elastic spacecraft dynamics affecting the guidance system, and to safety aspects of ground support equipment.

Traditional design criteria result in vibration loads being critical for most spacecraft structural elements. Structural design for dynamic loading conditions is complicated by the dependency of structural element loads on the response characteristics of the composite spacecraft which are, in turn, dependent on the design of the element. This requires an iterative approach to structural design and analysis. To implement this approach, the spacecraft is considered to consist of several structural subsystems, such as the primary structure and adapter, secondary structure and planetary science platform, propulsion system, superstructure and high-gain antenna, low-gain antenna and its damped support system, and the solar panels.

During preliminary design, loads will be assumed for each major element, based on past experience. As the preliminary design progresses, a mathematical model of each element will be developed. This model will then be combined with the mathematical models of other subsystems to obtain a modal analysis of the composite spacecraft structure. The early modal analysis will indicate areas of potentially deleterious coupling and may result in structural redesign. Response analyses will then utilize the latest modal analysis and assumed damping characteristics to provide loads for the next iteration. It is anticipated that several such iterations will take place as the spacecraft design evolves to improve mathematical models and to incorporate redesigned elements.

It should be noted that the modal combination computer program was first used on *Mariner* Venus '67 to analyze the tip-latched solar panel configuration. *Mariner* Mars 1969 is the first spacecraft program to utilize the modal combination computer program for a composite analysis of the entire spacecraft. Improved analytical methods for individual structural elements are also now available in the recently developed structural analysis and matrix interpretive system (SAMIS)[1] program, so that the structural analysis approach for *Mariner* Mars 1969 structural configuration is considerably modified from *Mariner* Mars 1964, particularly with regard to the tip-latched solar panel configuration and to the large planetary scan platform. To improve confidence in the spacecraft structural analysis, early modal tests are planned on a structural mock-up. Correlation of modal survey and analytical results will be made for major structural elements during assembly, and a modal survey of the composite spacecraft will be correlated with the composite analysis.

Response analyses may be checked by low-level forced vibration tests. This correlation will provide greatly increased confidence in the structural analysis results for the *Mariner* Mars 1969 configuration. The structural analysis goal is to eliminate the need for major structural redesign after the STM test, without using an excessively conservative design approach. Additionally, the results

[1]"Structural Analysis and Matrix Interpretive System (SAMIS) Program Report," JPL Technical Memorandum 33-307, September 1, 1966; and "Structural Analysis and Matrix Interpretive System (SAMIS) Program: Technical Report," JPL Technical Memorandum 33-311, November 1, 1966.

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of composite spacecraft analyses may be used in conjunction with launch vehicle analyses to provide a better definition of the structural loads and environments at the spacecraft interface.

It is believed that this combination of loads and criteria definition, composite dynamic analyses, structural design and stress analyses, and structural testing will result in a spacecraft with a high structural reliability, and in addition, a better basis for predicting subsystem loads and environments.

4. Electronic Parts Engineering

Electronic Parts Engineering has the responsibility for the *Mariner* Mars 1969 parts control program (electronic and electromechanical parts only), with the objective of control over part selection and acquisition. Specifically, the project goals are the use of the highest reliability parts available within the constraints of the schedule, and the maintenance of the part-type count at a total of 300. Attainment of these goals requires standardization of part types to achieve a high degree of commonality among the *Mariner* Mars 1969 subsystems controlled buying and part screening.

Following initial design review, all subcontractors and JPL cognizant engineers submit preliminary parts lists, tabulating each different part/application/documentation "combination" intended for use on each subsystem. Parts Engineering then reviews these lists recommending, where applicable, use of alternate parts because of higher qualification and reliability or for the purpose of standardization.

All attempts are made to use parts already qualified and evaluated. Where a requested part has not been qualified previously and no suitable qualified replacement is available, the part becomes a candidate for qualification or evaluation to whatever level is practical.

The procurement of "preferred parts" to preferred parts specifications is required where appropriate; as an alternative, the highest available level of commercial/military part is purchased and 100% electrically screened.

JPL Parts Engineering provides failure analysis support wherever necessary. When the subcontractor performs his own analyses, the results are reviewed by JPL.

5. Electronic Packaging Engineering Support

Electronic packaging engineering support functions include a broad spectrum of supporting tasks for the *Mariner* Mars 1969 spacecraft system organization and the spacecraft subsystem areas. Included are electronic packaging design, methods and processes, material and hardware selection for electronic equipment, electromechanical engineering, manufacturing processes, connector and wire selection, and high-voltage design support. The flight and flight-associated equipment responsibilities of the electronic packaging discipline are covered by the electronic packaging portion of the structure subsystem; included are the design, qualification and fabrication responsibilities for electronic assemblies, chassis, subchassis, and the shipping and handling equipment required for electronic equipment.

Electronic packaging design support covers spacecraft electronic and electromechanical equipment. The *Mariner* Mars 1969 Electronic Packaging Functional Requirements Document (M69-3-220) and the associated design, control, and process documents have been generated. The subsystem areas are supported in varying degrees to meet the needs of the individual cognizant engineers.

The connector and wire support area develops and/or qualifies the preferred connector, wire, and accessory products for project use. Assistance is also provided to the flight-hardware responsible areas in the procurement and screening of the required connector and wire products.

Method and process support includes a wide range of process documentation supporting the various electronic packaging design techniques and configurations utilized for the *Mariner* Mars 1969 spacecraft. The process documentation is divided into two major categories: (1) process specifications which give general engineering and quality assurance requirements, and (2) process bulletins which include detailed sequential process information. Technical areas covering the process documentation include material application, manufacturing, rework, assembly operations, and others applicable to electronic equipment.

E. Guidance and Control

1. Introduction

Guidance and Control Division analytical design efforts include functional design, analysis, and functional integration efforts related to the several subsystems of the *Mariner* Mars 1969 spacecraft for which the Guidance and Control Division is responsible. These subsystems include the attitude control, scan, central computer and sequencer (CC&S), power, and approach guidance demonstration subsystems. Work since the inception of the

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project has been concentrated in three areas: definition and negotiation of subsystem interfaces, system information-flow design for preflight and in-flight programming, loading, and testing of the CC&S subsystem, and participation in the design and integration of the approach guidance subsystem flight feasibility demonstration.

In the area of definition and negotiation of subsystem interfaces, functional requirements for the Guidance and Control Division subsystems have been reviewed, and telemetry and command assignments for the subsystems have been determined and negotiated. Work will continue in this area until all interfaces are frozen.

The CC&S subsystem utilizes a flexible, computer-oriented device in the spacecraft that is programmed to provide the desired sequencing and control. During system test and launch operations, semiautomatic paper-tape-controlled digital operational support equipment (OSE) accomplishes the loading and verification of the CC&S program and tests the CC&S to determine correctness of function and readiness for launch. After launch, modifications to the CC&S program may be made by radio command to accommodate changes in mission performance requirements.

Analytical design responsibilities include the system design of a data-handling and ground-based computer software system that can generate the necessary CC&S programs, OSE control tapes, and command messages necessary to utilize and test the CC&S before and during flight, as well as detailed software design for the portion of the data-handling system related to preparation of the OSE control tapes. As of this report, data system definition and design is in process, together with the development of implementation plans and schedules for producing the software. Nearing completion is definition and the preparation of a request for programming of a computer program to produce OSE control tapes from a description of the required test sequence and CC&S program.

The technology development and *Mariner* Mars 1969 flight feasibility demonstration of spacecraft-based optical approach guidance measurements involve both the sensor subsystem on the spacecraft and a software subsystem on the ground, part of which will be integrated into the Space Flight Operations Facility (SFOF) software system to demonstrate the feasibility of accomplishing the approach guidance process, utilizing spacecraft optical data on an operational time scale. The functional design of the software subsystem has begun and some of the interfaces with other SFOF programs, such as the Double-Precision Orbit Determination Program, have been examined. Programming support and schedule requirements are being developed. Planning for post-flight performance evaluation programs and other supporting software is also in process.

2. Central Computer and Sequencer (CC&S) Subsystem

The *Mariner* Mars 1969 CC&S effort was initiated in February 1966. The CC&S to be utilized is a new design oriented toward a special purpose computer capable of flight programming on the launch pad, and is flexible to allow in-flight reprogramming or updating as required. The phases of the over-all design and development include a JPL electrical design with subsequent packaging, fabrication, and test efforts subcontracted with the exception of the packaging associated with the magnetic core memory, which is being done by JPL.

The foundation for the new CC&S design is a combination of previous *Mariner* CC&S magnetic circuit techniques, of integrated circuit techniques resulting from advanced development applications, and of organization and performance techniques developed pursuant to the *Voyager* CC&S preliminary design by JPL. The major improvements inherent in the new design over previous CC&S designs are: (1) flexibility to change mission sequencing up to the time of launch and even in flight, if necessary; (2) flexibility to increase the number of sequenced commands late in the development cycle; (3) redundancy to assure that no single component failure mode could be mission catastrophic; (4) application of more recent state-of-the-art components to circuitry developed by up-to-date analysis techniques; (5) use of up-to-date packaging and fabrication techniques; and (6) functional capability (with an increase in memory storage) to handle more complex missions such as estimated for *Voyager*.

At present, the electrical design is 95% complete including worst-case analysis of all circuitry and the selection and specifications of all components to JPL standards. A breadboard has been fabricated and is presently undergoing temperature-voltage margin tests to verify analysis and other required performance capabilities.

The major subsystem contractor, Motorola Military Electronics Division/Western Center, was selected and the contract for the packaging design phase was initiated September 12, 1966. The design phase is proceeding on schedule and an RFP is presently being prepared for a contract to cover design documentation, fabrication, and test of a flight prototype CC&S.

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3. Spacecraft Power

Design studies of the power subsystem shown in the simplified block diagram (Fig. 4) have been made at JPL. TRW, Inc., has been selected for negotiation of the contract for this subsystem.

a. Solar panels. A solar panel configuration analysis has indicated that 4 rectangular panels, each measuring 35.5 in. wide $\times$ 84.2 in. long, and having a combined active area of 83 ft^2, will fit within the *Atlas/Centaur* launch vehicle's *Surveyor* shroud. Preliminary layouts are under way which have four and six electrical sections per panel. Layouts using both 1 $\times$ 2 and 2 $\times$ 2 cm N/P solar cells are being considered. New improvements in solar panel design such as multi-tab connected cells, insulation material, replacement modules, substrates (honeycomb or corrugated), and zener diodes are being examined. The new panels will not have solar vanes and will be tip-latched in the stowed position. Studies of potential solar cell suppliers are under way, and a contractor is expected to chosen in November.

b. Power conditioning equipment. Three switching booster regulator designs are currently being considered for *Mariner* Mars 1969. The designs have several basic differences. The first uses a method in which the pulse width modulation takes place in the base drive circuit (in front of the output transformer). The second approach uses a control technique that is similar to the first, except that the pulse-width modulation takes place in the secondary section of the output transformer. The third design uses a phase-shift technique to control the duty cycle and is more complex than the first two designs. Analysis and preliminary data indicate that the first design will have the highest efficiency.

The first and second booster regulator designs encountered problems in stability because the magnetic amplifier did not track properly. These problems have been solved and rough efficiency measurements on the units show 85–90%. The third design is in the breadboard stage and no performance data are available. A decision regarding which design approach to use will be made in early November.

The booster regulator fault detection and switching circuitry is being redesigned to utilize a new set of voltage detection limits. This was necessary because the output voltage of the booster regulator was raised to 56 v to overcome loop instability problems.

Work is progressing on the telemetry oscillator and transducers. A new oscillator is being designed which will utilize most of the old transducers, with minor modifications. A new battery charge current transducer with a double scale factor has been designed. The double scale factor improves the resolution of charge current monitor as the battery approaches full charge.

The battery charger/booster circuit design will be modified to incorporate improved components in the battery boost circuit. Specifically, the 2N1016 transistors will be replaced by 2N3846 transistors which are faster and have a higher collector current rating. A design is being considered that will locate the charger assembly in the power regulator. The charger control logic will be modified to accommodate a new battery charger/booster command mechanization.

The power stage of the main inverter has been designed, breadboarded, and checked out. For this design, two types of drive oscillators are being examined. One is a flux-type oscillator that will hold approximately $\pm 2\%$ accuracy. This oscillator has shown no problems when synchronized with the precision timing reference. The second is a unijunction oscillator that, with the selection of two components, may hold $\pm 1\%$ accuracy. The main inverter circuit will be ready for package design November 1.

The power synchronizer circuit design is almost complete. In the design, the precision oscillator uses a crystal for basic accuracy. The prime candidate for the application is the same 307.2-kHz, LDT-cut crystal, with minor package modifications, that was used on *Mariner* Mars 1964. There is a considerable amount of experience with this crystal from both the use and manufacturing points of view. A second candidate is a 38.4-kHz, NT-cut crystal with which there is much less experience but which would require three fewer divider stages to obtain the basic clock frequency of 2.4 kHz.

The frequency tolerance on the 307.2-kHz crystal is $\pm 0.01\%$ over the temperature range -10 to $+85$°C. There are less data available to support accuracy predictions using the 38.4-kHz crystal. However, the nominal frequency tolerance is $\pm 0.015\%$ over the same temperature range as above.

A power distribution system has been designed that will accommodate the latest command list. A breadboard of this design has been checked out and the schematic and parts list are ready for package design.

The 400-Hz inverters (single- and three-phase), and the 2.4-kHz maneuver inverter circuit designs are complete and ready for package design. However, the output transformer of the single-phase inverter is being modified to improve regulation.

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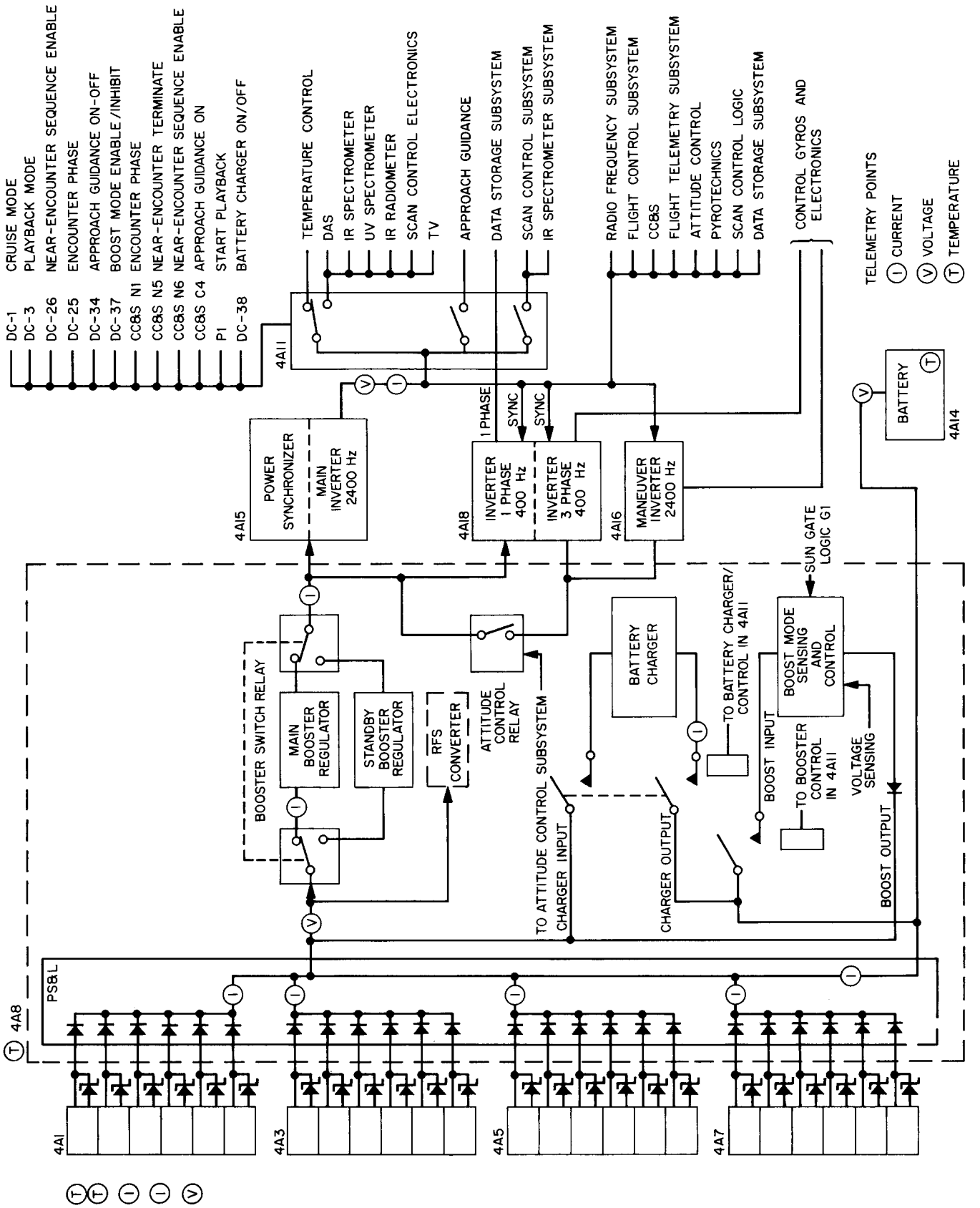

Fig. 4. Power subsystem simplified block diagram

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All units must be tested to determine the transient characteristic of the power subsystem.

c. Energy storage system. The energy storage system will be an 18-cell, sealed silver-zinc, alkaline battery. The normal capacity will be 43 amp-hr (1200 w-hr). Maximum charge and discharge currents are 2 and 15 amp, respectively. Voltage regulation will be from 25.8 v when discharged under load, to 35.2 v at the end of the 2-amp charge. Cell cases will be ABS plastic and the battery case will be ZK-60 magnesium. The battery will weigh 34 lb.

Battery procurement activities will begin in January 1967. The battery will be supplied to the subsystem contractor as government-furnished equipment. Improvements in the battery will be in the form of long-life terminal seals and separator systems.

4. Spacecraft Control

A contract has been let to Kearfott Division of General Precision for the procurement of gyros for the *Mariner* Mars 1969 attitude control. Negotiations have been completed on the Phase I Subsystems Contract for the attitude control and scan control with Honeywell, Inc. Personnel from Honeywell have reported to JPL under a letter contract and are assisting with modifications of the *Mariner* Mars 1964 attitude control design for *Mariner* Mars 1969 and with the development of the *Mariner* Mars 1969 scan control design.

The design of the electronic equipment bay for the gyro control, attitude control electronics, and CC&S is being modified from the *Mariner* Mars 1964 configuration to provide additional volume for the new CC&S. This modification requires changes in packaging of the gyro control and attitude control electronics units. This repackaging design for the gyro control unit is 85% complete and the design for the attitude control electronics package has been initiated.

Functional design of the attitude control and scan control is proceeding along with the spacecraft system design effort.

The Phase I contract for the approach guidance subsystem has been negotiated with Electro-Optical Systems and design effort has started under a pre-contract cost authorization. The approach guidance subsystem is being used in a *Mariner* Mars 1969 flight feasibility demonstration of approach guidance technology.

5. Operational Support Equipment

The operational support equipment (OSE) described here is the ground support equipment that provides the power, control, stimulation, monitoring, and interface simulation necessary for testing the spacecraft subsystems provided by the Guidance and Control Division. The attitude control, scan control, and approach guidance demonstration subsystems are supported by a single OSE unit, called the spacecraft control OSE. The CC&S subsystem and the power subsystem are supported by separate OSE units.

Each type of OSE is supplied in three test set configurations. The launch complex set (LCS) is a minimum-complexity configuration used to support the subsystem when it is installed on the spacecraft and the launch vehicle. The LCS is connected to the spacecraft subsystem that it supports only by lines in the umbilical cable. This set provides the functions necessary to power, command, monitor, and condition the subsystem during all launch pad operations. It is also used for spacecraft support during operations at the explosive-safe area and during dummy countdown practice. The system test set (STS) configuration is used to support its subsystem during system assembly and test operations, when the subsystem is installed in the spacecraft. To provide additional control, stimulation and monitoring functions, this test set is connected to the spacecraft subsystem by direct-access cables, in addition to the umbilical lines provided for the LCS. The LCS equipment is normally a part of the STS, in order to prevent the introduction of new spacecraft-OSE interfaces at the time the spacecraft is moved to the launch pad. The laboratory test set (LTS) is comprised of the STS and the additional equipment necessary to support testing of the spacecraft subsystem as a unit isolated from the spacecraft, in such situations as subsystem functional checkout and acceptance testing, and subsystem environmental testing. The LTS is connected to its subsystem through the lines normally used to interconnect the subsystem with other subsystems on the spacecraft (the spacecraft ring harness) as well as through the direct-access and umbilical lines used by the STS. The added functions provided by the LTS consist primarily of supplying power to the subsystem, simulation of interfaces with other spacecraft subsystems, and added stimulation and monitoring functions.

Because the *Mariner* Mars 1969 spacecraft is an adaptation and expansion of the *Mariner* Mars 1964 and *Mariner* Venus 67 spacecraft configurations, much of the OSE originally developed for the 1964 project, and later modified for the 1967 project, will be further modified for the 1969 project. The power OSE requires only modifications to the existing test sets. The spacecraft control OSE requires new assemblies for scan and approach guidance support, in addition to modifications of the existing attitude control OSE. New OSE is being designed and

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built to support the new computer-oriented CC&S configuration of the *Mariner* Mars 1969 spacecraft.

The design and fabrication of each type (CC&S, power, and spacecraft control) of OSE is combined with that of the flight subsystem in a single major subsystem contract. This arrangement minimizes the difficulties encountered in integrating the flight equipment and the OSE, and provides improved opportunities for test-team training during the functional and acceptance testing of the prototype and flight subsystems. The present status of each type of OSE is described in the following paragraphs.

a. Power operational support equipment. The *Mariner* Mars 1964 power OSE (P-OSE) design has been discussed in Vol. II of the following Space Program Summaries: 37-21, pp. 20–23; 37-23, pp. 27–30; 37-26, pp. 20–21; 37-28, pp. 16–17.

Design changes for support of the *Mariner* Venus 67 spacecraft have been limited to: the addition of over-voltage protection diodes to the spacecraft external power circuits; refinement of the over- and under-voltage interlock sensor to prevent issuance of a command to transfer the spacecraft to external power when the external power is set to an over-voltage condition at turn-on; the refinement of the battery power-on command circuit to provide a minimum command of 100 msec; an increase in wire size of a command line to increase the starting torque of the motorized switch used for transferring the spacecraft between internal and external power; the addition of a fan to the AC regulator assembly to reduce the output voltage drift; modification of the solar power simulator to improve its simulation of the characteristics of the solar panel with zener-diode regulation; and the replacement of a digital voltmeter with one that has improved common-mode noise rejection capability.

The P-OSE modifications for *Mariner* Mars 1969 will include new voltage level detectors for monitoring the spacecraft booster-regulator, new circuits to test the spacecraft power switching, replacement of P-OSE dummy loads for the spacecraft booster-regulators and inverters, addition of equipment to test the synchronizer oscillator, separation of the LTS adapter console control panel into two panels and assemblies, and modification of battery charger test command circuits.

Initial negotiations with a contractor are in process.

b. CC&S operational support equipment. The CC&S for the *Mariner* Mars 1969 spacecraft is a new computer-oriented configuration that differs significantly from the timer-oriented designs of past *Ranger* and *Mariner* Projects. Development of the new OSE required to support this CC&S design has been aided considerably by experience and equipment developed during the now-cancelled *Mariner* Mars 1966 Project and during the early stages of the *Voyager* Project.

In 1964, work was begun on OSE to support the development of a time-oriented CC&S for a *Mariner* Mars 1966 spacecraft. One of the objectives of the OSE development was to investigate the application of semiautomatic equipment controlled by digital logic and a program stored on paper tape. The data processing logic of the test set was designed to incorporate considerable flexibility, so that the set could be adapted to the testing of other subsystems by changes in the interface circuits and the program stored on the control tape. When the *Mariner* Mars 1966 program was cancelled, development of the CC&S and its OSE was completed under advanced-development funding. The adaptability of the OSE design was demonstrated when a set of OSE was required to support the development of an experimental computer-oriented CC&S for the *Voyager* Project. A second set of OSE, using the same control logic but modified interface circuits, was designed and built in a few months. Experience gained during testing of the *Voyager* computer-oriented CC&S equipment with this set indicated the desirability of certain changes to the control logic, which have been incorporated into the *Mariner* Mars 1969 OSE design. The *Mariner* Mars 1966 test set described above has since been equipped with modified interface circuits and used to support the initial development of the *Mariner* Mars 1969 CC&S, which is derived from the *Voyager* computer-oriented design. The OSE prepared for testing the *Voyager* equipment has been utilized in the breadboard testing of the *Mariner* Mars 1969 OSE design.

The LTS and STS configurations of the *Mariner* Mars 1969 CC&S OSE are identical, consisting of eleven assemblies mounted in two 6-ft racks. The LCS omits two of the manual control assemblies that will not be required in the launch area. Each of the test sets can be operated in either a manual control or a semiautomatic mode. In the manual-control mode, all interfaces between the OSE and CC&S can be controlled by front-panel switches, and monitored by front-panel displays of lights or meters. In the semiautomatic mode, the necessary control instructions and monitoring criteria are punched on a paper tape that is read by the OSE as it conducts the test. The digital logic of the OSE causes the printing of a record of the test, and permits the OSE to advance automatically from one test step to the next as long as the status information returned from the CC&S to the OSE matches the monitoring criteria punched into the tape. The set will automatically stop and print an error message if incorrect information is returned; the operator can then

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return the OSE to the manual-control mode for trouble shooting. The preparation of the control tapes will be accomplished by an off-line computer. (The development of the software for this preparation process is described in the Guidance and Control Section of this SPS.)

The digital logic of the *Mariner* Mars 1966 OSE was comprised entirely of standard *nand* gates composed of diodes, transistors, and other discrete circuit elements. The *Mariner* Mars 1969 OSE design continues to use *nand* logic, but has has been converted to the use of integrated circuits. The electrical design of the 1969 OSE has been completed by JPL, and fabrication of a breadboard to verify and test the design is 50% complete. This breadboard will later be integrated with the CC&S breadboard, and then will be used for initial testing of the prototype CC&S being packaged and built by the subsystem contractor, Motorola. An engineer and a technician from the contractor are presently in residence at JPL to assist in breadboard fabrication and testing.

The contract with Motorola was initiated in September 1966. Since that time, Motorola has completed an initial review of the OSE design and is now preparing to fabricate a prototype of the OSE.

c. Spacecraft control operational support equipment. The attitude control OSE for *Mariner* Mars 1964 spacecraft, as modified for support of the *Mariner* Venus 67 spacecraft, forms the nucleus of the spacecraft control OSE for *Mariner* Mars 1969. The 1964 design has been described in Vol. II of the following Space Programs Summaries: 37-21, pp. 25–28; and 37-28, pp. 13–16.

Changes to support the *Mariner* Venus 67 Project include a new light-stimulus hood for the Canopus sensor, the addition of controls for this hood to the equipment racks, a change in the cruise Sun sensor limit detector to accommodate a change in the voltage limit level, the addition of stimulus circuits and hoods for the planet and terminator sensors, and minor changes in buffering circuits.

F. Telecommunications

The four telecommunications subsystems, all based on their *Mariner* Mars 1964 counterparts, are in the early stages of design/development for the *Mariner* Mars 1969 missions. Work is being done by JPL cognizant engineers and their respective contractors. A number of studies in system performance, trade-offs, and design questions are under way; preliminary design reviews have been completed. The following subsystem information is in accordance with present *Mariner* Mars 1969 project requirements and may change as a result of the studies mentioned above.

The radio frequency subsystem will incorporate major modifications to the 1964 design in that two traveling-wave-tube power amplifiers will be used, and a single spacecraft bay will house the flight equipment. The high-gain antenna will be repositioned to suit the 1969 flight geometry. A change in the low-gain antenna's radiation pattern is being studied, as well as certain design improvements in the transponder and operational support equipment (OSE). The contract, with Philco's Western Development Laboratory, was initiated during the previous report period.

The telemetry subsystem consists of the data encoder on the spacecraft and appropriate decoding equipment on the ground, as well as OSE. The major change aboard the spacecraft from the 1964 subsystem is the addition of a 66⅔-cps data rate; on the ground, the 1964 ground telemetry subsystem hardware will be used on the *Mariner* Mars 1969 missions to demodulate the telemetry. However, unlike the 1964 system, decommutation will be performed by the DSIF's telemetry and command processor (TCP). The TCP is designed as mission-independent equipment to be used with mission-dependent software. Multiple bit rate capability extending beyond 66⅔ cps, and block coding are being studied. The contract, with Texas Instruments, Inc., was executed in the latter part of this report period. Additional studies are being carried out at JPL on the trade-offs between use of a computer-based bit synchronizer versus the *Mariner* Mars 1964-type PN synchronization technique.

The command subsystem consists of the command detector and decoder aboard the spacecraft, the ground command subsystem, and OSE. The command capability, in terms of usable commands, is essentially that of the 1964 design. Automatic acquisition of the command subcarrier received by the spacecraft is being studied, as is the use of the TCP (with a mission-dependent modulator) for command generation. The contract to Motorola, Inc., was executed in September.

The data storage subsystem, used to record high data rate encounter-science for playback to Earth, consists of an analog tape recorder, a digital tape recorder, and an associated analog-to-digital converter. The total design-goal capacity is 1.87×10^8 "equivalent bits" of storage. The recorders are both two-speed, four-track, endless-loop machines resembling the *Mariner* Mars 1964 recorder in design principle. The subsystem contractor is Texas Instruments, Inc., with Lockheed Electronics as

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subcontractor for the tape recorder mechanisms. This contract also was executed in late September.

G. Propulsion

1. Propulsion Subsystem

The *Mariner* Mars 1969 spacecraft propulsion subsystem will provide the necessary impulse to remove or reduce injection dispersion errors so that a Mars flyby with a sufficiently small miss distance can be reasonably assured.

It is planned that the propulsion subsystem will be very similar to that used on the *Mariner* Mars 1964 spacecraft. The propulsion subsystem utilizes a liquid monopropellant, anhydrous hydrazine, as the propellant. It is functionally a constant-thrust rocket with propellants pumped by pressure-regulated gas. Principal components consist of a high-pressure gas reservoir, a pneumatic pressure regulator, a propellant tank and bladder, valves and a rocket engine. The rocket engine contains a catalyst to maintain the decomposition of hydrazine which is initiated by injecting a small amount of nitrogen tetroxide (contained in the oxidizer start cartridge) into the thrust chamber during start-up. The hypergolic reaction of the nitrogen tetroxide and hydrogen heats the catalyst to a temperature that will sustain the monopropellant reaction.

Because the development of spontaneous catalyst has progressed to such a great degree at this time, the use of this type of catalyst is being considered for the *Mariner* Mars 1969 propulsion subsystem. The utilization of a spontaneous catalyst could provide improved reliability by eliminating the requirement for a bipropellant start and, as a result, the oxidizer start cartridge and its associated valves and plumbing. Also this catalyst should improve the minimum velocity capability of the subsystem.

The effort to date has been concentrated on the preparation of a bid package to be included in a request for proposal (RFP) for the fabrication of the *Mariner* Mars 1969 propulsion subsystem. The RFP was released to industry November 1, 1966.

2. Pyrotechnic Subsystem

The *Mariner* Mars 1969 spacecraft pyrotechnic subsystem consists of squib-actuated devices and the electronic control units which initiate the squibs.

For the spacecraft pyrotechnic devices, the changes from the *Mariner* Mars 1964 design are:

(1) Addition of the release devices required for spacecraft separation. In *Mariner* Mars 1964, these devices were part of the launch vehicle systems. It is planned to use *Mariner* Mars 1964-type release devices, which will be initiated (fired) by the launch vehicle.

(2) Addition of squib-actuated valves for activation of the infrared spectrometer cool-down gas supply. It is planned that the squibs and valves will be functionally identical to those used in the *Mariner* Mars 1964 propulsion subsystem.

(3) Utilization of 1-amp/1-w no-fire squibs in the pin-pullers. A change in squib type is required because of the nonavailability of the type used on *Mariner* Mars 1964; the change to a 1-amp/1-w no-fire type is considered to reduce certain risks, including those in the area of system reliability, for *Mariner* Mars 1969 and future missions.

(4) Revision of the number of pin-pullers, unlatching events, and the number of pin-pullers per event, as determined by the requirements of the system design.

For the pyrotechnic control units, it is planned to retain the two electronic subassemblies which redundantly perform the functions of energy storage (with charged capacitor banks) and squib firing (with SCR switches that are triggered by external command).

Some changes in component types and circuit logic configuration may be proposed for reliability improvement. Changes in the circuit will be required for compatibility with new or changed requirements, dictated by the system design.

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III. *Voyager* Project

A. Introduction

1. Objectives

The primary objective of the *Voyager* Program is to carry out scientific investigations of the solar system by instrumented, unmanned spacecraft which will fly by, orbit, and/or land on the planets. Emphasis will be placed on acquisition of scientific information relevant to the origin and evolution of the solar system; the origin, evolution, and nature of life; and the application of this information to an understanding of terrestrial life. The primary objective of the *Voyager* missions to Mars, beginning in 1973, is to obtain information relative to the existence and nature of extraterrestrial life; the atmospheric, surface, and body characteristics of Mars and the planetary environment by performing automated experiments on the surface of, and in orbit about, the planet. A secondary objective is to further our knowledge of the interplanetary medium between the planets Earth and Mars by obtaining scientific and engineering measurements while the spacecraft is in transit.

2. Project Plan

All *Voyager* missions will be conducted as events of an integrated program in which each individual flight forms a part of a logical sequence in an over-all technical plan of both lander and orbital operations. The *Voyager* design will provide for the carrying of large scientific payloads to the planet, the telemetering of a high volume of data back to Earth, and long useful lifetimes in orbit about the planet and/or on the planetary surface. Hardware will be designed to accommodate a variety of spacecraft and/or capsule science payloads, mission profiles, and trajectories. Particular emphasis will be given to simple and conservative designs, redundancy wherever appropriate, and a comprehensive program of component, subsystem, and system testing.

Over-all direction and evaluation of the *Voyager* Program is the responsibility of the Office of Space Science and Applications (OSSA) of the National Aeronautics and Space Administration (NASA). Management of the *Voyager* Project and implementation of selected systems is the responsibility of the Jet Propulsion Laboratory (JPL) of the California Institute of Technology.

3. Technical Description

Planning shall be based upon launching two identical planetary vehicles by a single *Saturn V* launch vehicle

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at each Martian opportunity, starting with the 1973 opportunity. Each planetary vehicle is to consist of a flight spacecraft, flight capsule bus, and surface laboratory, with science experiments conducted in 1973 from the orbiter, during descent and on the planetary surface. The flight spacecraft with its capability for a 400-lb science subsystem (250 lb of instruments) will weigh aproximately 2,500 lb; its retropropulsion subsystem may additionally weigh up to 15,000 lb; and the total flight capsule (capsule bus and surface laboratory) will weigh up to 5,000 lb. The flight spacecraft will be a fully attitude-stabilized device utilizing celestial references for the cruise phase, and will be capable of providing velocity increments for midcourse trajectory corrections and for Mars orbit attainment by both the flight spacecraft and the flight capsule. On-board sequencing and logic and ground command capability will be provided. The flight spacecraft will supply its own power from solar energy or from internal sources and will be capable of maintaining radio communications with Earth. In addition, the flight spacecraft will be thermally integrated and stabilized and will monitor various phenomena near Mars and during transit, and will telemeter this information back to Earth; it will monitor and telemeter data pertaining to spacecraft operations. The flight spacecraft will also provide the flight capsule with services such as power, timing and sequencing, telemetry, and command during the transit portion of the missions and may serve as a communications relay. The sterilized flight capsule bus will be designed for separation from the flight spacecraft in orbit, for attaining a Mars impact trajectory, for entry into the Martian atmosphere, for descent to the surface, and for soft-landing the surface laboratory, which will have surface lifetimes of up to 2 days in 1973 and later goals of as much as 6 mos. The flight capsule bus will contain the propulsion, guidance, control, communications, and data handling systems necessary to complete its mission, while the surface laboratory will contain all power, sequencing, and communications for the conduct of surface operations.

No deep-space flight tests of the flight spacecraft are planned, and compensation will be made by more extensive ground tests to the extent possible. The test program for the flight capsule, which is planned to include Earth-entry flight tests, will investigate entry dynamics.

During the capsule Phase B time period, assessment shall be made of the quantitative effect of various growth factors on useful landed weight including increases in launch vehicle performance, increase in minimum Martian surface pressure, and availability of current weight contingency.

4. Sequence of Events

The Project plan is being implemented as follows:

The Spacecraft System procurement activity (Task C) is continuing with the three Phase B fixed-price contracts for the accomplishment of mission, system, and management studies. One system contractor will be selected, through competition, for award of a CPAF Phase C contract and, subsequently, a multiple-incentive CPIF-type Phase D contract.

The procurement action for the total capsule system includes selection of multiple Phase B contractors in the last quarter of FY 67 and subsequently, from among these, one Phase C and D contractor for the capsule bus system will be selected to carry out, essentially in parallel, Phases C and D of the spacecraft system procurement. The surface laboratory Phase C procurement will be open competition for one contractor who will carry out Phases C and D.

B. Telecommunications

1. Engineering Data Compression Evaluation Console

a. Introduction. Described in SPS 37-39, Vol. II, is an implementation study for a spacecraft engineering data-handling system using data-compression techniques. The report climaxed a research effort that attempted to establish criteria for compression of engineering data and, further, attempted to evolve a system philosophy.

As a result of the favorable conclusions reached during this effort, a program of experimentation has been initiated which involves the procurement of a data-compression evaluation console (DCEC) to test the performance of the various system algorithms when used in conjunction with a zero-order predictor for data redundancy reduction.

b. DCEC description. The DCEC (Fig. 1) is organized as a piece of laboratory test equipment. It is designed for maximum flexibility commensurate with usefulness and budget allocations. The input medium will normally be digital magnetic tape read in by an integral tape system. Complete flexibility is available in the selection of input data since each file on a tape can be either a hybrid or strictly digital computer simulation of peculiar or complex mode event periods of an unmanned spacecraft mission. Fourteen data channels will be simulated for each run.

Provision is also made for a digital simulator to be used as an input medium for certain experiments calling for all channels to process the same waveform.

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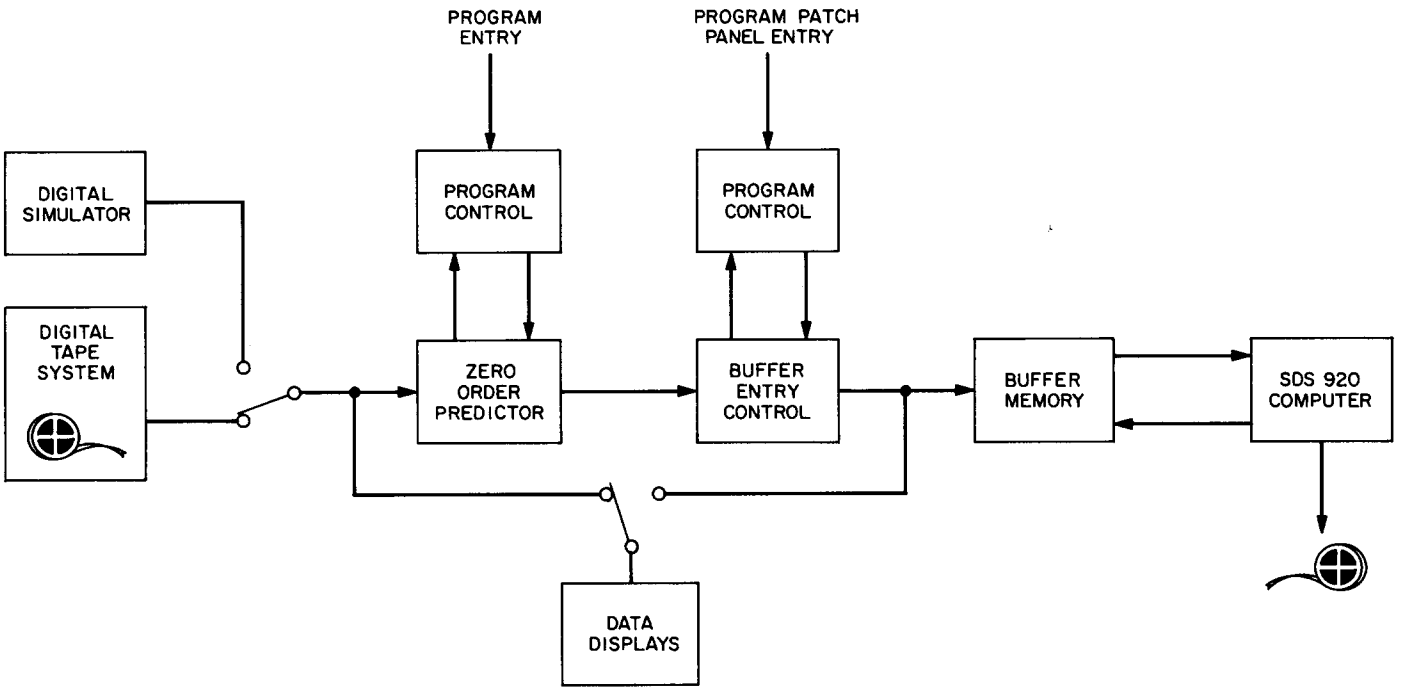

Fig. 1. DCEC block diagram

The input data will be fed to a zero-order predictor where significance of each data point is determined by employing a programmed control parameter. The required programmed control parameter is an aperture code separately programmed for each channel. The aperture establishes how much a given sample value must deviate in amplitude from the last significant sample taken from that channel in order to be considered significant.

If a sample is considered significant it is sent to the buffer memory input control circuitry where the volume of data in the system buffer memory is monitored. The sample will be loaded into the buffer only if the criterion for the buffer entry control algorithm is satisfied.

The DCEC can be programmed (by patch panel) to operate the buffer entry control circuitry using any one of five algorithms:

(1) Automatic reduction of a programmed input sampling rate, relative to buffer fullness

(2) Automatic expansion of the significance test aperture, relative to buffer fullness

(3) Per channel allotment of total buffer volume, relative to present buffer fullness

(4) Combination of 1 and 3

(5) Combination of 2 and 3

The various control parameters that vary the degree to which each channel is affected by a given algorithm are individually programmed for each channel. The programming flexibility is so broad that, if desired, the buffer entry rate of one channel may not be affected at all, while another may be drastically cut back as the buffer fills.

The system output data is read out upon command into an SDS 920 computer which will act as an output formatting and recording device. The data transfer rate will be programmed to simulate the relative rate of a spacecraft communications link.

Various additional checkout features are also included in the DCEC, such as bypass of the zero order predictor or the buffer entry control logic, or both.

Important information to be gained from experiments with the DCEC includes: 1) the effects of data compression on various types and combinations of actual spacecraft system data waveforms; and 2) to what extent retrieval of necessary information is impaired by buffer limitations or buffer queuing routines.

2. 50-w Spacecraft Traveling Wave Tube Amplifier Development

a. Introduction. A 50-w traveling wave tube amplifier (TWTA) is being developed for *Voyager* spacecraft use. The TWTA consists of a 50-w traveling wave tube

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(TWT), a power supply capable of operating from 25 to 50 v dc input and an RF output filter. The TWT, power supply, and RF filter will be packaged within a pressurized container to prevent high-voltage breakdown and to allow operation during a launch or entry pressure profile.

b. Traveling wave tube development. Four experimental TWTs have been constructed. All have experienced vacuum seal, thermal, or cathode difficulties. The most valid design data were generated by Serial No. 102, and are summarized in Table 1.

Table 1. Engineering model TWT 102 test results

Helix voltage, v	Beam current, ma	Basic efficiency, %	Output power, w	Saturated gain, db	Helix current, ma
2150	90	24.2	46.7	16.2	12
2250	100	25.0	56.3	16.7	15
2250	110	25.5	63.1	18.0	20
2300	120	25.6	70.8	20.9	28

The original design called for a TWT with 22% beam power to RF power conversion efficiency (basic efficiency) and a collector operated at less than 50% of the helix voltage (depressed collector). This was expected to give 43% over-all efficiency. Difficulty has been experienced in obtaining proper operation with a depressed collector. TWT 102 was limited to a voltage depression of 25–30% below the helix voltage by the rapidly increasing helix interception current shown in Fig. 2.

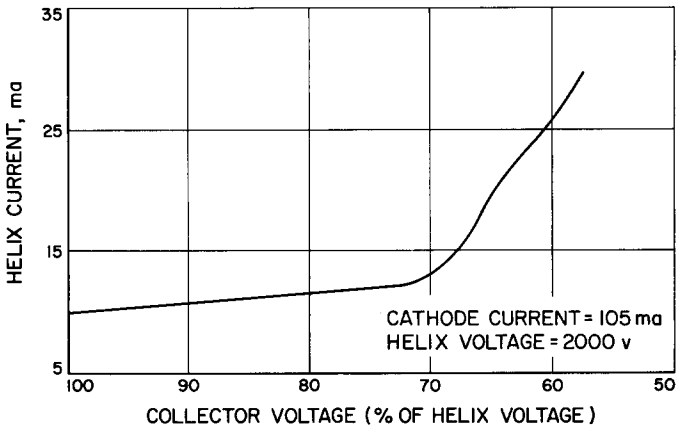

Fig. 2. Helix current versus collector voltage

Even though a basic efficiency of 25.6% has been achieved (Fig. 3) only 31.3% over-all efficiency resulted, as shown by Fig. 4.

Saturated gain in the TWTs fabricated has been low. As shown in Table 1, the maximum saturated gain of

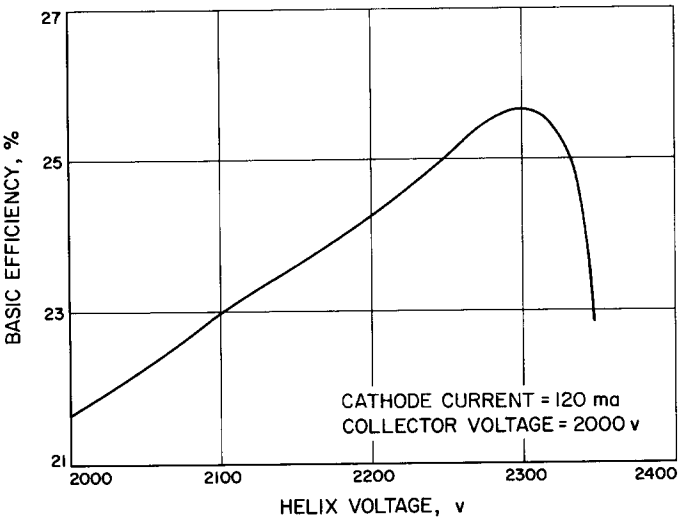

Fig. 3. Basic efficiency versus helix voltage

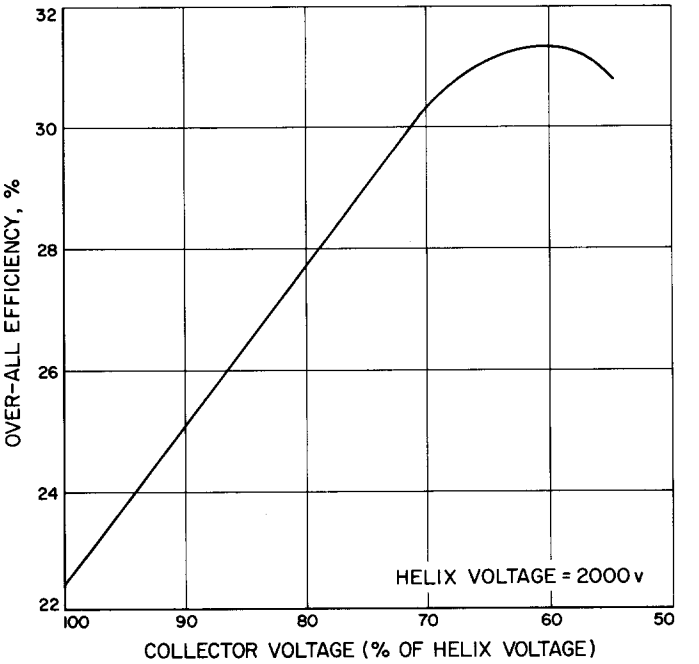

Fig. 4. Over-all efficiency versus collector voltage

TWT 102 was 20.9 db. The specification in effect at that time required 24 db of saturated gain. However, this requirement has recently been increased to 30 db as the result of a study which showed that better over-all transmitter efficiency would result if the TWTA required less drive.

Further development is being made to further increase basic efficiency, reduce interception current, and increase the gain.

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Film-oxide cathodes were used in the early experimental TWTs. The design calls for sprayed-oxide cathodes, but spray jigs were not available and it was felt that data on the RF portion of the TWT could be gained by making this substitution during the early part of the development. Some of the difficulties experienced with interception current are felt to be the result of poor performance by the film cathodes. Spray jigs for sprayed-oxide cathodes have been received recently and all future experimental TWTs are to utilize sprayed-oxide cathodes. The electron gun of TWT 102 has recently been replaced with a sprayed-oxide cathode to measure the effect on interception current.

It was felt that slowing of the electron beam toward the collector end of the tube might be a contributor to the interception current problem. A TWT has been fabricated with a modified periodic-permanent magnet focusing array. In this tube the period of the magnetic field is varied along the length of the TWT to match the plasma wavelength of the electron beam as the beam is slowed due to the RF interaction process under saturated drive conditions. This TWT has not been tested to date.

c. Power converter development. Power converter design was essentially complete in early October. As shown in Fig. 5, efficiency was within specification limits at −10 and 25°C. No valid data have yet been taken at 75°C. Ripple sensitivity was calculated to be out of specification for some ripple frequencies, as shown by Fig. 6.

Ripple sensitivity is determined by the ratio of power supply output ripple to input ripple. Ripple sensitivity is shown in Fig. 6. Zero db in this figure refers to the JPL specification limit. Additional filtering was added. No further test data have yet been generated.

Recently it was determined that the manufacturer of the power transistor, on which the design was based, had decided to stop production for economic reasons. A redesign effort has accordingly begun. Another transistor type must be selected and the circuit design will be modified to compensate for any difference in transistor parameters. The effect of this change on the over-all development has not yet been determined.

d. Packaging. The TWTA will be packaged in a hermetically sealed stainless steel enclosure filled with sulfur hexafluoride (SF_6) gas. Considerable attention has been given means of detecting leaks that might occur during preflight handling. An internal pressure transducer was considered, and means for detecting SF_6 external to the container were investigated. It was found that a mass spectrometer available at JPL could readily detect the presence of a leak by detecting the presence of the SF_6 leaking from inside the container. SF_6 is not normally present in the atmosphere in significant amounts. Sufficient space will exist within the enclosure so that a flight program could add a pressure transducer to the design, if desired.

No significant detail design can be accomplished on the TWTA enclosure until the TWT and power converter mechanical designs are firm.

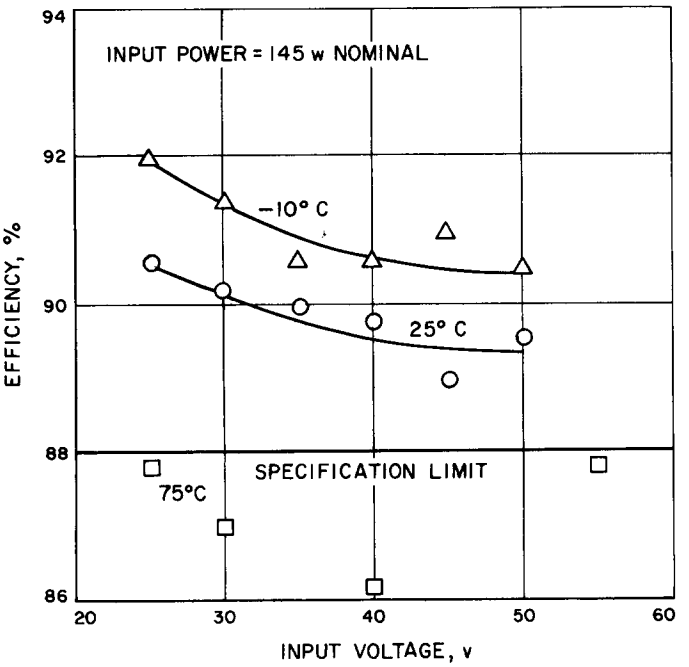

Fig. 5. Power converter efficiency versus input voltage

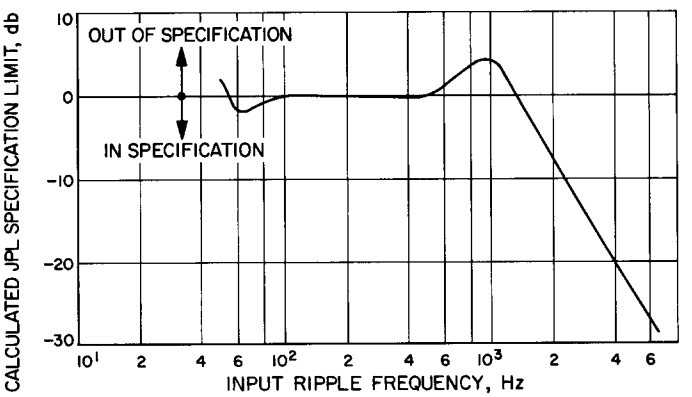

Fig. 6. Audio susceptibility versus input ripple frequency

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IV. Future Projects

The Future Projects Office of the Laboratory is responsible for the direction, sponsorship and, in cooperation with other elements of the Laboratory, the origination of advanced mission and future project studies in the areas of interest and applicability to the Laboratory.

The objective of this effort is to insure that the preliminary technical information and plans for the various potential missions, together with the associated information on spacecraft systems, instrumentation systems, launching vehicle systems, resources, and schedule be available as required. This preliminary mission study information will be used primarily for defining specific flight mission programs, and as generalized guidance for the Laboratory research and advance development programs.

Each study conducted by the Future Projects Office has been, and will continue to be, documented in a formal final report. This section of the Space Programs Summary is allocated to provide an abstract or summary of the results of each study including, where appropriate, reference to the final report for additional details.

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